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AN INVESTIGATION OF A THERMAL ICE-PREVENTION SYSTEM FOR A C-46 CARGO AIRPLANE

III - DESCRIPTION OF THERMAL ICE-PREVENTION EQUIPMENT FOR WINGS, EMPENNAGE, AND WINDSHIELD

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

ADVANCE RESTRICTED REPORT

AN INVESTIGATION OF A THERMAL ICE-PREVENTION SYSTEM FOR A C-46 CARGO AIRPLANE

III - DESCRIPTION OF THERMAL ICE-PREVENTION EQUIPMENT FOR WINGS, EMPENNAGE, AND WINDSHIELD

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SUMMARY

As a part of an investigation of a thermal ice-prevention system for a Curtiss-Wright C-46 airplane, the equipment for the wings, empennage, and windshield has been designed, constructed, and installed by the Ames Aeronautical Laboratory of the National Advisory Committee for Aeronautics. The research was undertaken in order to provide the C-46 airplane with protection from ice formations and to continue the general development of thermal ice-prevention equipment.

This report is the third of a series on the C-46 project and describes (1) the revisions to the wings, empennage, and windshield for thermal ice prevention, (2) the temperature- and pressure-measurement equipment installed in the airplane for the determination of the performance of the system, and (3) the results of static-load structural tests of a specimen of the wing outer-panel leading edge which indicated that the strength had not been impaired.

INTRODUCTION

This report is the third in a series on a comprehensive investigation of a thermal ice-prevention system for a C-46 cargo airplane. The first two reports of the series are presented as references 1 and 2. The NACA has designed and installed satisfactory thermal ice-prevention systems for the Lockheed 12A airplane, the Consolidated XB-24F airplane, and

the Boeing B-17F airplane (references 3, 4, 5, and 6), and this investigation is a continuation of the development which was initiated in those earlier researches.

A description of the construction and installation of the thermal ice-prevention system specified in references 1 and 2 in a production model of the C-46 airplane, as well as a description of the instrumentation installed to determine the performance of that system, is presented in this report. Also reported are the results of static-load tests of a portion of the wing outer-panel leading edge which were conducted to investigate the effect of the installation of the ice-prevention system on the strength of the structure.

The investigation was undertaken at the Ames Aeronautical Laboratory of the NACA, Moffett Field, Calif., in cooperation with, and at the request of, the Air Technical Service Command of the U. S. Army Air Forces. The appreciation of the NACA is extended to the Curtiss-Wright Corporation for the valuable consulting and liaison services of Messrs. Harold Hirsch and Francis Doyle, and for granting permission to publish the results of the structural tests which were performed by Curtiss-Wright personnel on the wing specimen.

DESCRIPTION OF THERMAL ICE-PREVENTION EQUIPMENT

General

The C-46 airplane (fig. 1) is a twin-engine low-wing monoplane of the heavy cargo type powered by two Pratt and Whitney Model R-2800-51 engines having a sea-level rating of 2000 horsepower each.

The general arrangement of the airplane as revised for thermal ice prevention is shown in figure 2. The ice-prevention system is based upon the transfer of heat from the engine exhaust gas to air which is then caused to flow along the inner surface of any portion of the airplane for which protection is desired. In the application of this principle to the C-46 airplane, the necessary heat is removed from the exhaust gas by means of two exhaust-gas-to-air heat exchangers installed in each nacelle, as shown in figure 2. The heated air from the outboard exchangers is used to provide ice protection for the wing outer panels and tips. The inboard exchangers supply heated air for heating of the cockpit and for the prevention of ice formations on the wing inboard panels, empennage, and windshield.

The design of the thermal ice-prevention system as installed on the C-46 was based on flight conditions of maximum range at 18,000 feet pressure altitude and 0° F ambient-air temperature. An analysis of the heat requirements for the wings, empennage, and windshield (reference 1) indicated that the required capacity for each of the four exhaust-gas-to-air exchangers would be about 300,000 Btu per hour with an airflow rate of 4130 pounds per hour and a temperature rise of 300° F.

Primary Heat Exchangers

The exhaust-gas-to-air heat exchangers installed on the airplane are of the flat-plate cross-flow type and are shown in figure 3. They were fabricated from sheets of Inconel pressed into the required shape and welded together along the edges. At the forward end of the exchanger, all the plates are welded together and also to the exhaust-gas inlet. At the rear of the exchanger, however, the two outer plates are not attached to the inner plates or to the gas outlet, thus allowing the inner core to expand independent of the outer cooler plates. The gas outlet is attached to the inner group of plates. The plates, approximately 9 inches by 15 inches, are assembled to form 17 air passages 0.21 inch wide and 16 exhaust-gas passages 0.28 inch wide. A more complete description of the exchanger, the design analysis, and the results of preliminary flight tests are presented in reference 2.

The heat-exchanger installation is shown in figure 4. The forward end of the exchangers was rigidly attached to the engine mount by a yoke-type clamp; therefore, a universal joint was required between the exhaust-gas collector rings and the heat exchangers to compensate for relative motion between the collector rings and engine mounts. The exchangers were supported at the rear end with a channel-and-lug arrangement which allowed the exchanger inner plates and exhaust outlets to expand rearward for about 1/4 inch. The revised collector ring and the exchanger supports for the right outboard installation are shown in figure 5. The air-inlet scoops and outlets attach directly to the exchangers, as shown in figures 4 and 6. Valves for the purpose of directing the heated air to the ice-prevention system or discharging it to the free stream were installed immediately after the exchanger air outlets. The discharge valves and outlets were fabricated from stainless steel as a precautionary measure in the event of exchanger failure. The heated-air ducts from the discharge valves to the leading edge of the wing were fabricated of aluminum and, in the case of both sides

of the left nacelle, were formed as venturi meters in order to provide means for evaluating the rate of air flow.

The shrouding of the exchanger installation is shown in figures 7, 8, and 9. The entire exchanger assembly up to the wing leading edge is covered by a single fairing containing outlets for the exhaust-gas and heated-air discharge. A cooling-air system for the universal ball joint was incorporated in the design in order to prolong the service life of the joint. This system is shown diagrammatically in figure 7 and consists of a shroud around the inner side of the ball joint forming a 1-inch annular space, between the shroud and the joint, through which cooling air is circulated. The cooling-air inlet scoop and discharge are formed from one piece and are shown in the completed installation (figs. 7 and 9).

Heated-Air-Distribution System

The heated-air-distribution system is shown in figure 2. No blowers are installed in the primary-air system, the circulation of the air depending upon the ram available at the exchanger air inlets and the pressure conditions at the discharge locations. In the case of the outboard exchangers, the air is admitted to the wing outer-panel ice-prevention system at a point just outboard from the nacelle. For single-engine operation of either engine, the heated air can be divided between the right- and left-wing outer-panel systems by operation of crossover valves located as shown in figure 2.

In the case of the inboard exchangers, the heated air passes through the leading edges of the inboard panels in 6-inch-diameter ducts to a common junction on the left side of the fuselage. A portion of the air is diverted from the ducts in the inboard panels to supply the inboard-panel leading-edge ice-prevention system. From the junction in the fuselage, the heated air is directed forward through a 4-inch duct to the secondary heat-exchanger installation in the airplane nose and aft through a 6-inch duct to the empennage. The straight portions of the distribution ducts were formed from 0.020-inch-thick 24S0 alclad aluminum alloy, and the elbows were formed from 0.032-inch-thick 3S aluminum alloy. All of the foregoing ducts, with the exception of the inboard-panel duct, were thermally insulated with a layer of 1/32-inch-thick asbestos paper and a layer of 1-inch-thick material of the rock-wool type. The ducts were then covered with aircraft linen.

Several valves, usually of the butterfly type, were located throughout the heated-air supply system in order to

control the distribution of the air flow. The locations of some of these valves are shown in figure 2 and the remainder will appear in subsequent figures. The means of valve control will be discussed later in the report.

Wings

A typical section of the wing outer-panel leading edge as revised for thermal ice prevention is shown in figure 10. In order to promote the transfer of heat from the internal air to the airfoil skin, the air is caused to circulate against the inner face of the skin by the application of a double-skin system. The inner skin extends rearward from the leading edge to the 10-percent-chord point, and is corrugated in such a manner that a series of chordwise air passages are formed between the inner and outer skin. The heated air enters the chordwise passages through a gap at the leading edge between the upper and lower segments of corrugated inner skin.

Spanwise distribution of the heated air is obtained by means of a duct formed by the installation of a spanwise baffle extending from the upper to the lower corrugated skins and located at approximately 5.5 percent chord. One of the factors influencing the location of the baffle was the fact that the wing main ribs terminated at about 5.5 percent chord and the leading-edge loads were carried by nose ribs forward of that point. The overlapping of the existing main and nose-rib webs precluded the installation of a continuous baffle and, also, the nose-rib webs prohibited spanwise flow of the heated air. The revised design, therefore, incorporated a new set of nose ribs designed to provide a spanwise air passage and a continuous baffle which was easily fabricated and installed. The nose-rib loading is transmitted to the main ribs, or to intermediate nose-rib stiffeners located between the main ribs, through an extruded angle-and-bolt arrangement on each side of the baffle.

The left-wing outer-panel leading edge with the inner corrugated skin and revised nose ribs installed is shown in figure 11. The nose ribs are located as follows: one at wing station 11, 15 inches apart from stations 22 to 232, 30 inches apart from stations 232 to 292, and none outboard of station 292. (For wing station locations, see fig. 2.)

A rear view of the baffle plate installed in the wing leading edge is shown in figure 12. The L-shape stiffeners

shown are located behind the intermediate nose ribs and attach to the spanwise hat-section member located on the upper surface of the wing (fig. 10). The stiffeners shown midway between the L sections are also located behind nose ribs and are attached to the wing main ribs as shown in figure 13. The extruded angle bolted through the baffle plate to an identical angle on the nose rib, and the end of the corrugated inner skin, beveled to provide a larger exit area, may also be seen in figure 13.

The necessity for the addition of the nose-rib liner (fig. 10) to the system was indicated in tests in which air was blown into the revised leading edge before attachment to the wing. The tests, discussed in reference 1, showed that the nose liner was required in order to obtain a spanwise distribution of the heated air which conformed to the distribution found necessary by the design analysis. The nose liner extends spanwise from wing station 11 to the last nose-rib location, station 292. Heated air for the corrugation passages leaves the nose-liner duct through 1-inch-diameter holes spaced approximately 4 inches apart on the top and bottom surface of the liner near the baffle plate (fig. 10). The air then passes through a plenum region between the nose liner and corrugated skin and into the chordwise passages through the leading-edge gap. The 1-inch holes in the nose liner were located near the baffle rather than at the leading edge mainly because of installation difficulties experienced in slipping the liner into position. A view of the nose liner installed in the leading edge of the right wing is shown in figure 14.

After leaving the corrugation passages, the heated air circulates throughout the wing interior, passing through reinforced holes in the spar webs, and is discharged through the aileron and flap slots.

The revisions to the wing tips are shown in figure 15. The type of leading-edge construction is similar to that employed in the case of the wing outer panel with the exception that the corrugated inner skin is replaced by a dimpled type which is more adaptable to the double-contour forming necessary in the wing-tip revisions. The baffle plates in the wing outer panel and tip are located at the same chord station at the tip joint, thus extending the spanwise distribution duct to the extreme end of the wing tip.

The thermal ice-prevention system for the leading edge of the inboard panels is shown in figure 16. The corrugated

inner skin, in this case, extends in a continuous sheet from 5 percent chord on the wing lower surface to 10 percent chord on the upper surface. The heated air for the system is obtained from the supply ducts from the inboard exchangers through the installation of slide valves on the upper surface of the ducts. The heated air is caused to flow through the corrugation passages, from bottom to top, by surrounding the supply duct with a baffle. The web of the inboard-panel front spar is highly stressed and therefore no holes were cut for circulation of the heated air through the interior of the wing. The air returns to the free stream through existing openings in the structure. The inboard-panel leading-edge skin, with corrugated inner skin attached, is shown during installation on the wing in figure 17. The supply duct from the left inboard exchanger and the angles with which the baffle is attached to the ribs can also be seen in figure 17.

Empennage

The general arrangement of the thermal ice-prevention equipment in the empennage of the C-46 airplane is shown in figure 18. Distribution of the heated air between the stabilizer and the fin, and between the right and left sides of the stabilizer, is controlled by adjustment of the three butterfly valves shown in figure 18.

The revisions to the stabilizer and the fin leading edges are identical and a typical section is shown in figure 19. The design is similar to that employed in the case of the wing outer panel. The corrugated inner skin extends rearward to 10 percent of the airfoil chord and the corrugation passages are identical to those in the wing. A rear view of the baffle plate and corrugated inner skin for the stabilizer leading edge is shown in figure 20. The completely revised leading edge prior to installation on the stabilizer is shown in figure 21. After leaving the leading-edge region, the heated air circulates through the stabilizer or fin interior and is discharged at the elevator or rudder slot. The dimpled-skin and continuous-baffle type of construction employed in the wing-tip revisions was also utilized in the case of the stabilizer and fin tips (figs. 15 and 18). The modified leading edge of the fin tip is shown in figure 22 during construction. The baffle has been bent away from its attachment angles in order to show a larger portion of the dimpled skin and the air gap at the leading edge. The revised leading edge prior to installation on the fin is shown in figure 23.

Secondary Heat Exchanger and Windshield

A secondary heat exchanger was installed in the nose of the airplane to provide heated air for cockpit and windshield heating as shown in figure 24. A schematic drawing indicating the flow of air through the installation is presented in figure 25. Secondary air was used for the heating medium in preference to primary air in order to minimize the possibility of carbon-monoxide poisoning of the flight crew. The secondary exchanger is of the all-aluminum, flat-tube type similar to current intercooler designs. The design analysis of the windshield-heating requirements (reference 1) indicated that the secondary air would not have sufficient heat capacity for windshield protection, and therefore provision was made to discharge the primary air from the secondary exchanger over the outer surface of the windshield for additional heating. This was accomplished by attaching a secondary skin to the fuselage surface immediately forward of the windshield. The gap between the fuselage and secondary skins is about 1/4 inch. Primary air from the secondary exchanger is introduced into this space through slots in the fuselage skin and is discharged at the base of the windshield through an opening 1/4 inch by 58 inches along the lower edge of the windshield. An additional portion of the primary air is bypassed around the exchanger and employed to heat the regions surrounding the secondary-air inlet and passing light. The secondary air enters the system at the airplane nose and, by means of the valves located immediately behind the exchanger, the pilot and copilot can control the mixture temperature of the cockpit ventilating air. When windshield heating is desired, the heated secondary air from the exchanger can be directed to the blower by operation of the two valves shown in section A-A of figure 25.

The internal thermal ice-prevention system for the windshield is shown in figure 26. The installation for each side of the windshield consists of an inner removable panel spaced 3/16 inch from the existing outer panel and an air-distribution duct along the bottom of the windshield. After passing through the gap between the inner and outer panels, the heated air is discharged into the cockpit at the top of the inner panel. Previous experience with similar air-heated-windshield installations has shown that ice tends to form on the unheated center post and extend over the windshield area. In the present design, therefore, heating of the center post was provided by the installation of a removable channel which formed a chamber for circulation of the heated air as shown in figure 26.

Controls

The valves for the operation and regulation of the thermal ice-prevention system will be discussed in the order of (1) those controllable in flight, and (2) those adjusted on the ground and fixed for flight. The four discharge valves at the exchanger air outlets are operated by electric motors which are controllable from the cockpit. The motors are located inside the nacelles adjacent to the valves and the two are connected by a short linkage mechanism through a cut-out in the nacelle skin. The control motor installed in the outboard side of the left nacelle is shown in figure 27. The rated torque capacity of the motors is 70 inch-pounds. A circuit diagram for the installation of the motors is presented in figure 28. Operation of the switch labeled "heat to wing" and "discharge" in the figure causes the motor to drive the valve from one position to the other. The motor is stopped at each limit of the valve travel by a limit switch mechanism attached to the valve arm. The green and amber lights shown in figure 28 indicate the position of the discharge valve, amber designating the discharge condition. The control motors as supplied by the manufacturer were equipped with high-temperature override units, but these devices were not connected into the system for the preliminary performance tests. The discharge-valve control-motor panel, as installed in the cockpit of the airplane, is shown in figure 29.

The crossover valves (fig. 2), the four valves at the secondary heat-exchanger outlet (figs. 24 and 25), and the windshield heated-air supply valve (fig. 25) are adjustable in flight by means of push-pull cable controls. The crossover valve controls are located in the main cabin of the airplane and the remaining controls are within reach of the pilot and copilot. All of the remaining valves of the system are normally fixed prior to flight; although a few, such as the inboard-panel slide valves (fig. 16), can be adjusted in flight if necessary.

DESCRIPTION OF INSTRUMENTATION

Temperature- and pressure-measuring apparatus was installed in the thermal ice-prevention system for the purpose of determining the performance of the equipment. The principal factors to be measured and upon which the evaluation of the performance is based may be taken as (1) weight rate of heated-air flow to all parts of the system, (2) temperatures

of the heated air, and (3) temperature rise of components of the airplane exposed to the heated air. The air-flow rates are determined by venturi meters installed in the supply ducts and the temperatures are determined with thermocouples.

An index to the pressure-orifice and thermocouple locations for the entire thermal ice-prevention system is shown in figure 30, and the specific locations are given in figures 31 to 38. The dash numbers following the thermocuple and pressure-orifice designations in figures 31 to 38 indicate the type of mounting as detailed in figure 39. The pressure-orifice locations, for the most part, were confined to the venturi meters, although a few orifices were located in the air inlets of the left-nacelle heat exchangers, in the exhaust-gas stack of the left outboard exchanger, and in the secondary-exchanger windshield installation. The single pressure orifice located in the exhaust stack immediately forward of the left outboard exchanger (fig. 31) was connected to a manifold-pressure gage. All of the remaining pressures, including reference total and static pressures from a pitot-static head located beneath the airplane near the nose, were connected to a glass-tube alcohol manometer shown in figure 40.

All of the thermocouples were of iron-constantan wire with the exception of one exhaust-gas chromel-alumel thermocouple (designated G1, fig. 31) located immediately forward of the left outboard heat exchanger. The iron-constantan thermocouples (about 200 in number) were connected to a switching arrangement which provides manual or automatic thermocouple selection. When the unit is operated manually, the leads from the swithcing assembly are connected to a temperature-indicating instrument and the operator selects and observes the temperature of any one of the thermocouples. For automatic operation, the switching assembly is driven by a small electric motor and the leads are connected to a recording galvanometer. The temperatures are recorded as deflections on a 2.4-inch-wide roll of photographic film. The switching unit consists of nine banks of 24 thermocouple points each, and automatic operation is obtainable as follows: (1) Record the 24 points of any one bank and then stop; (2) record any one bank continuously; (3) record all nine banks in order and then stop; and (4) record all nine banks in order continously. The temperature-selecting and -recording unit is shown in figure 41. The calibrating panel shown beside the temperature recorder is used to impress a series of constant known voltages on the recorder film for calibration purposes. The chromel-alumel thermocouple was provided with

leads at the observer's station which can be connected to an indicating millivoltmeter or potentiometer.

The accuracy of the data obtained from the thermocouple installation used to measure skin temperatures (type 1, fig. 39) has been difficult to establish. In order to compare the data from this type of thermocouple with a more delicate but probably more accurate type of installation, eight special thermocouples, designated as surface thermocouples, were installed at wing outer-panel station 159 at the locations shown in figure 34. The surface-type thermocouples were made from 0.005-inch-diameter manganin and constantan wires which were butt-welded together and then rolled to a flat strip approximately 0.002 inch thick. The thin strip was then placed on the outer surface of the airfoil skin with the junction within 1/4 inch of the washer (type 1) installation. Because the eight surface thermocouples were manganin-constantan, they could not be added to the iron-constantan recorder circuit and therefore they were connected to a switch with a separate set of lead wires.

STRUCTURAL TESTS OF WING LEADING EDGE

General

The test specimen which was fabricated for the purpose of investigating the strength of the revised wing leading edge is shown in figures 42, 43, 44, and 45. The leading-edge contour of the specimen for the entire span was taken as that existing at wing station 22 (chord = 191.7 in.) and may be considered, with negligible error, as an NACA 23017 profile. The span of the specimen (60 in.) incorporated three main ribs and two intermediate ribs. After fabrication in the shops of AAL, the specimen was tested at the Buffalo, N.Y., Airport Plant of the Curtiss-Wright Corporation, Airplane Division. The test data presented herein are reproduced from a Curtiss-Wright report. The tests were conducted by Messrs. J. Kline and M. J. Berman under the supervision of Mr. E. L. Dashefsky, C-46 Structures Project Leader, and were made available for inclusion in this report through the courtesy of the Curtiss-Wright Corporation.

Test Procedure

The loads to be applied to the specimen were determined by adjusting the initial limit loading established for the

Curtiss-Wright model 20 to meet the present requirements. The initial loads were increased by a factor of 1.25 to account for an increase in gross weight from 40,000 to 50,000 pounds and by a factor of 1.3 to allow for reduced yield strength in the structure as a result of elevated temperatures. The revised limit loads were therefore established as 1.25 × 1.3 = 1.63 times the initial limit loads. The ultimate loads were taken as 1.5 times the revised limit loads. The results of these load calculations are summarized in table I. The unit ultimate loads from table I were plotted to give a curve of chordwise loading. The areas under the curve were measured and centroids located from which the concentrated ultimate loading (shown in fig. 46 for 15 in. of span) was established.

The loads were applied to the specimen with the lever and whiffletree arrangement shown in figures 47 and 48. The steel mounting angles at the aft end of the specimen (fig. 42) were bolted to the test frame and all loads were applied and indicated with the three turnbuckle-dynamometer systems shown in figure 48. Upper-surface pull points were located at the center of each of the four rib bays, and the load was distributed over four corrugations through a steel-channel and wood-block installation as shown in figure 49. Distribution of the compression loads was effected with a steel-channel and felted-board arrangement which may be seen in figure 47.

Several Ames deflection dials were located around the specimen at the points shown in figure 50. These dials were attached to a wood frame which in turn was secured to the two steel channels at the rear of the specimen in order to minimize errors caused by deflections of the test frame. The loads were applied to the specimen in increments of 20 percent of the revised limit loads. For each test configuration initial loadings of 10 percent of the revised limit loads were applied to the specimen and the readings of the deflection dials were recorded. These readings were arbitrarily established as "zero" set. Subsequent to the application of loadings equal to or greater than 60 percent of the revised limit loads, the loads were reduced to the arbitrarily chosen loading of 10 percent of the revised limit loads and readings of the deflection dials were again recorded. A comparison of these readings with the zero-set readings established the set, or permanent deformation, produced by each load application. Five different configurations of the specimen were tested, the final form being completely without nose ribs as shown in figure 51. The alterations to the specimen for each

test, the locations of the dial deflection gages, the test results, and remarks are summarized in table II.

Comments

The deflections and sets for tests 1, 4, and 5 are presented in figures 52, 53, and 54, respectively. The deflections for tests 2 and 3 were approximately the same as for test 1 and are not presented. The test results indicate no appreciable change in maximum deflection until all the nose ribs were removed for test 5. From a structural standpoint the specimen was satisfactory in all tests since all deflection curves up to the limit load were straight lines, indicating that no yielding had taken place. The deflections of test 5, however, might be unsatisfactory from aerodynamic considerations. Of the five configurations tested, the number 4 arrangement is recommended for future use, although consideration should be given to the possible further weight reduction to be obtained through the use of thinner sheet for the fabrication of some parts of the system.

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TABLE I.- DETERMINATION OF ULTIMATE UNIT LOADING FOR STATIC TESTS OF C-46 WING OUTER PANEL LEADING-EDGE TEST SPECIMEN

Percent chord	Unit pressures (lb/sq in.)					
	Upper surface			Lower surface		
	Initial limit	Revised limit	Ultimate	Initial limit	Revised limit	Ultimate
0.5	0.65	1.06	1.58	0.54	0.88	1.32
2.0	1.30	2.12	3.18	.50	.81	1.21
5.0	1.52	2.47	3.70	.36	.59	.88
7.5	1.50	2.44	3.66	.29	.47	.70
10.0	1.44	2.35	3.52	.22	.36	.54

TABLE II.- RESULTS OF STATIC-LOAD STRUCTURAL TESTS OF SPECIMEN OF C-46 AIRPLANE WING OUTER-PANEL LEADING EDGE

Test	Model configuration	Deflection dials installed	Remarks
1	As shown in figures 42, 43, 44, and 45	1, 2, 3, 4, 5, and 6	At about 130 percent limit load several spot welds attaching baffle plate to stiffeners of main end ribs failed. Test continued to ultimate load. Spot welds replaced with 1/8-inch rivets.
2	Removed spanwise "Z" stringer from lower-surface corrugations (fig. 45), replaced intermediate nose-rib stiffeners (fig. 45) with 5/8- by 5/8- by 0.032-inch 24S-T alclad angles riveted to baffle. Aft clip angles attaching intermediate nose-rib stiffeners to nose ribs (detail a, fig. 44) replaced with 24S-T alclad bent clip angles. Forward legs of upper and lower corrugated baffle angles (fig. 42 and detail a, fig. 44) removed.	1, 2, 3, 4, 5, and 6	Deflections recorded approximately the same as for test 1. Deflections and maximum set at ultimate load were less than test 1 because of rivets in place of spot welds. Slight buckling of skin aft of hat section at ultimate load and also slight buckling of baffle-plate stiffener angles.
3	Removed nose-rib-reinforcing angles (fig. 43) and plugged holes with 1/8-inch rivets. Added dial 7 to indicate magnitude of buckling aft of hat section.	1, 2, 3, 4, 5, 6, and 7	Dial at new location (7) recorded 0.004 inch (max.) at ultimate loading. Deflections and sets did not increase appreciably.
4	Removed the two intermediate nose ribs (fig. 45) and plugged holes with 1/8-inch rivets. Dial 6 read same as 1, so was moved to location 8.	1, 2, 3, 4, 5, 7, and 8	Up to limit load, all curves were straight lines, indicating no yielding had taken place.
5	Removed remaining nose ribs and plugged holes (fig. 51).	1, 2, 3, 4, 5, 7, and 8	No permanent buckling at ultimate load. Marked increase in deflections.

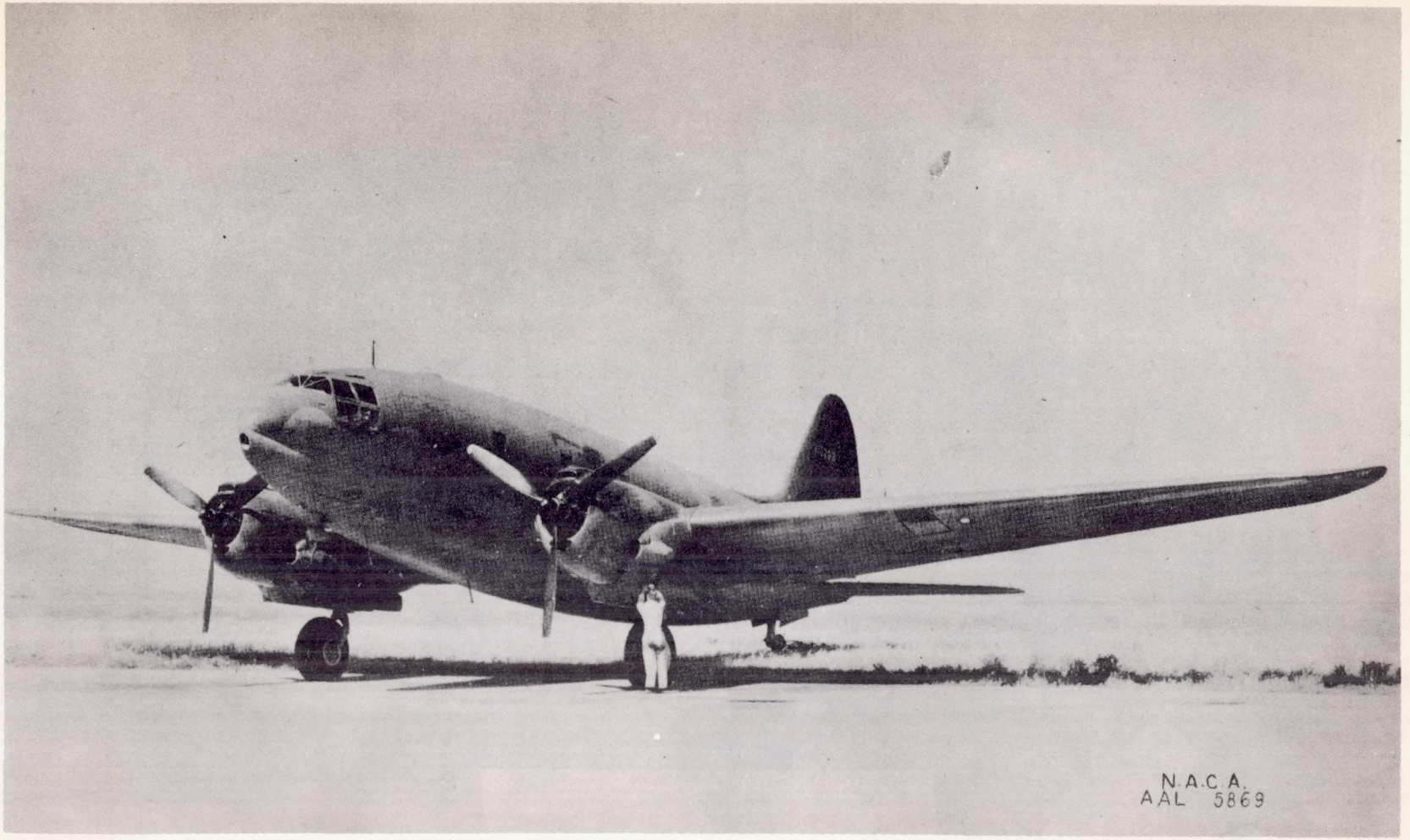

Figure 1.- The C-46 airplane for which the NACA designed and installed thermal ice-prevention equipment for the wings, empennage, and windshield.

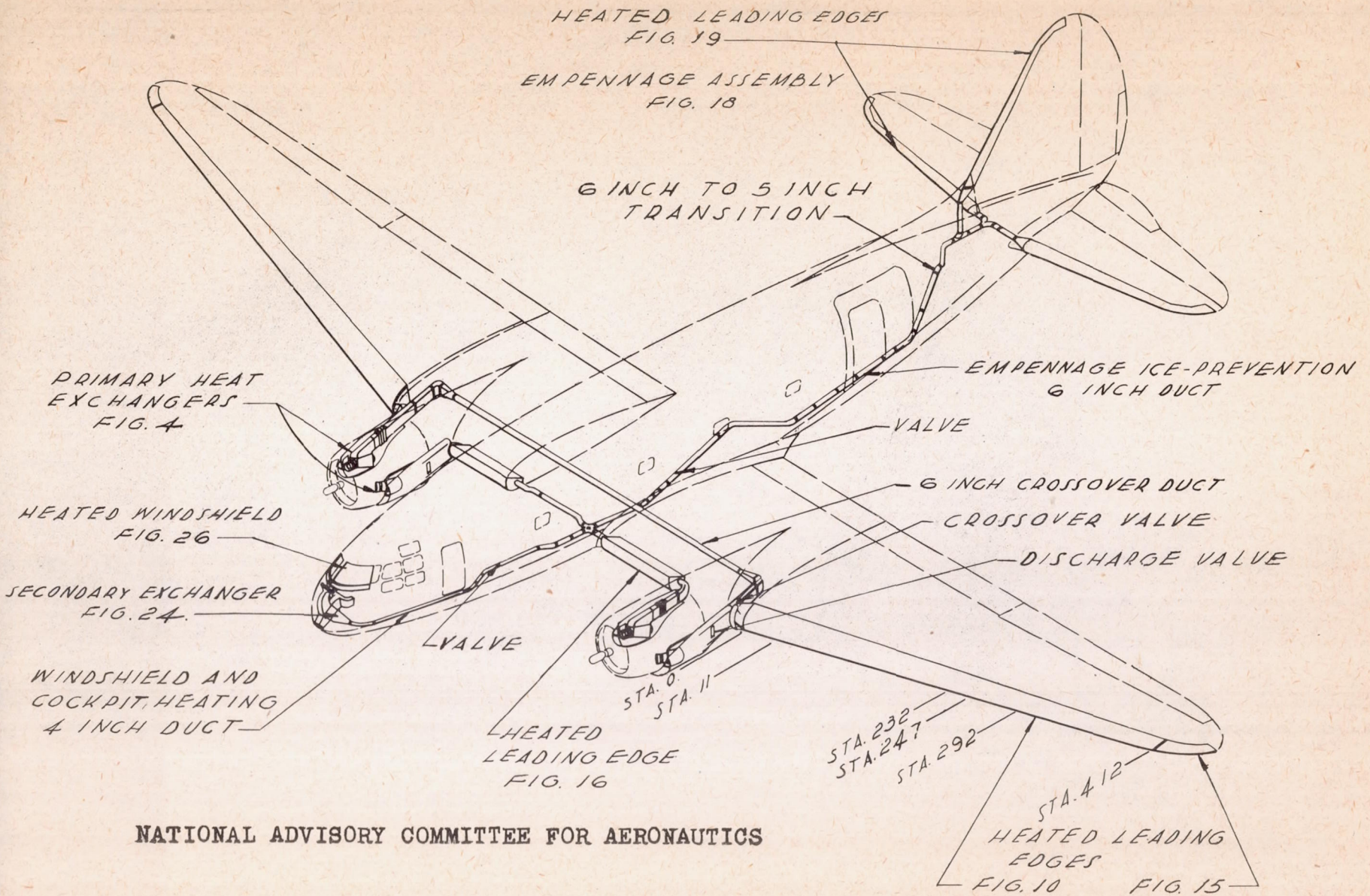

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Fig. 2

FIGURE 2- GENERAL ARRANGEMENT OF THERMAL ICE-PREVENTION EQUIPMENT IN C-46 AIRPLANE

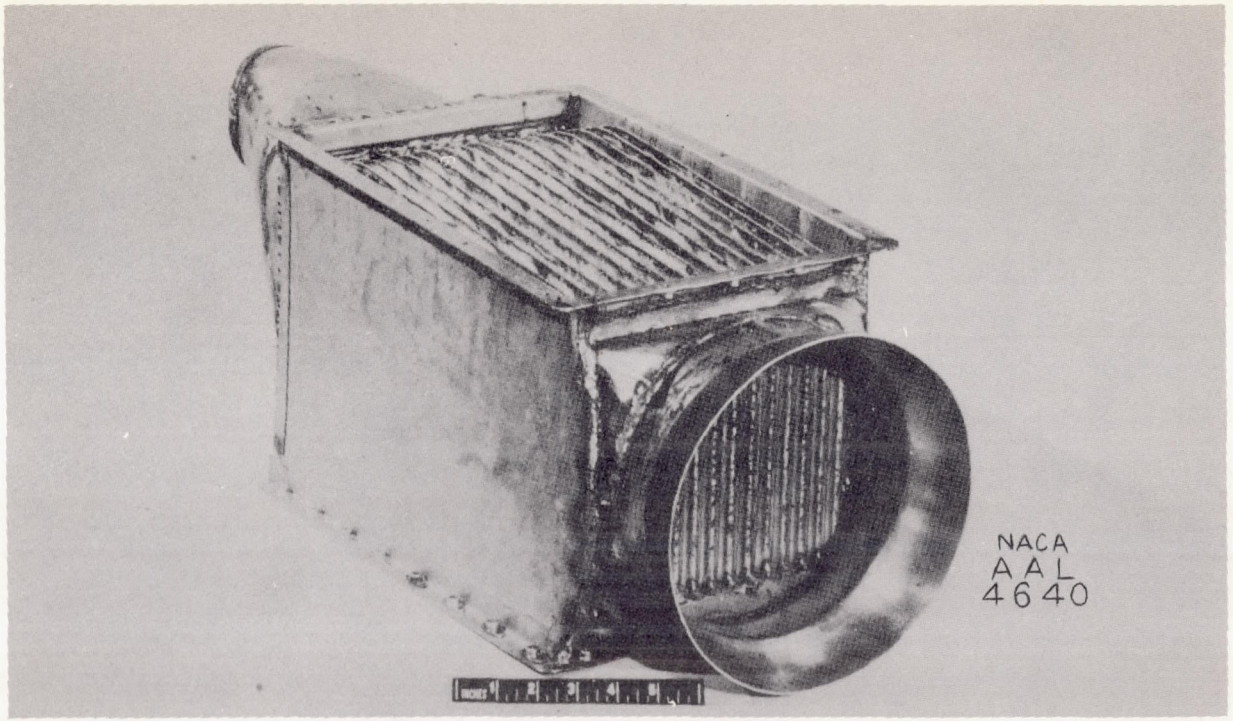

Figure 3.- Cross-flow plate-type exhaust gas-air heat exchanger as installed on C-46 airplane viewed from exhaust inlet side.

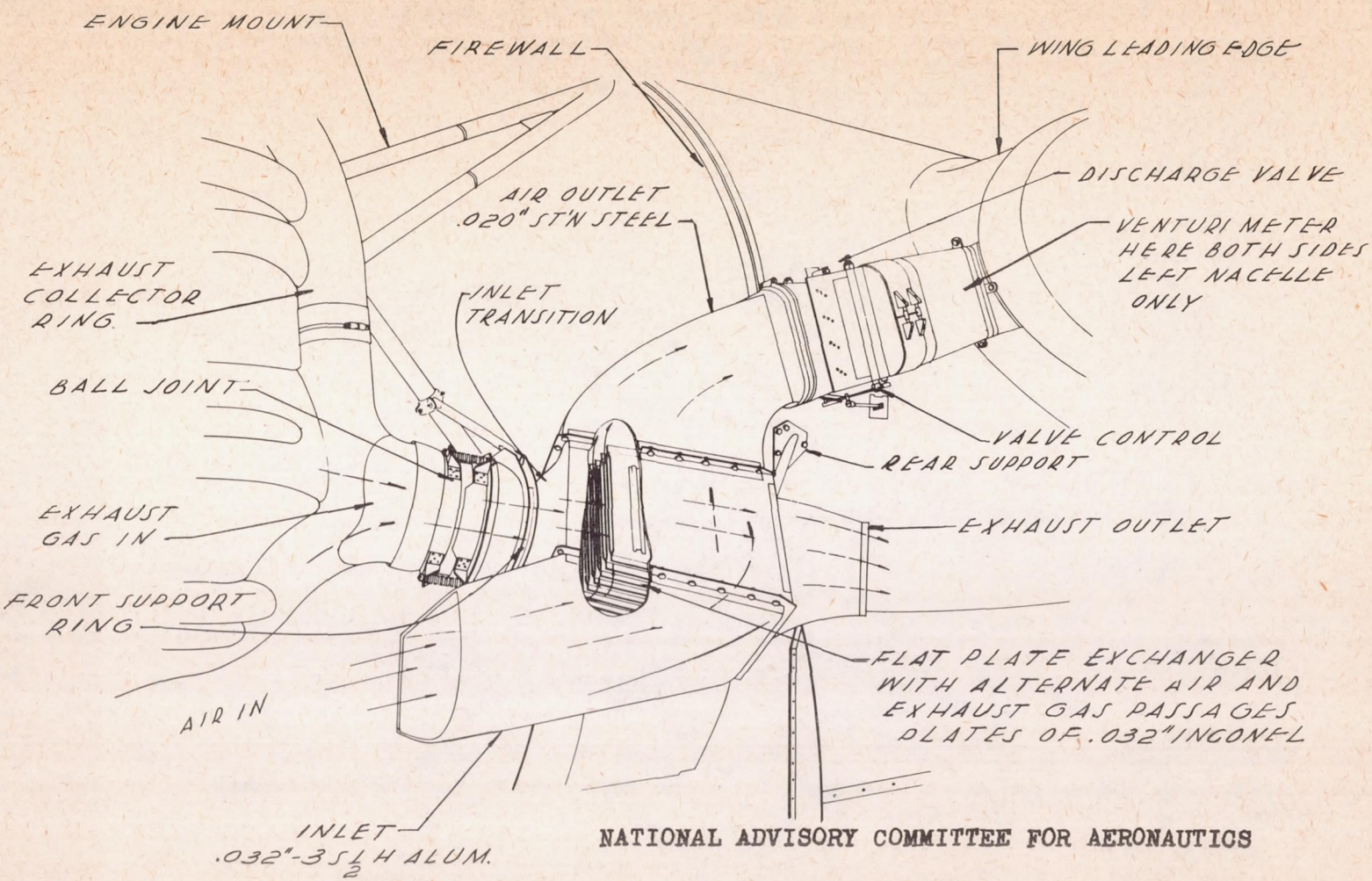

FIGURE 4:— DETAILS OF HEAT-EXCHANGER INSTALLATION ON C-46 AIRPLANE SHOWING PATHS OF EXHAUST GAS AND AIR FLOW

Fig. 4

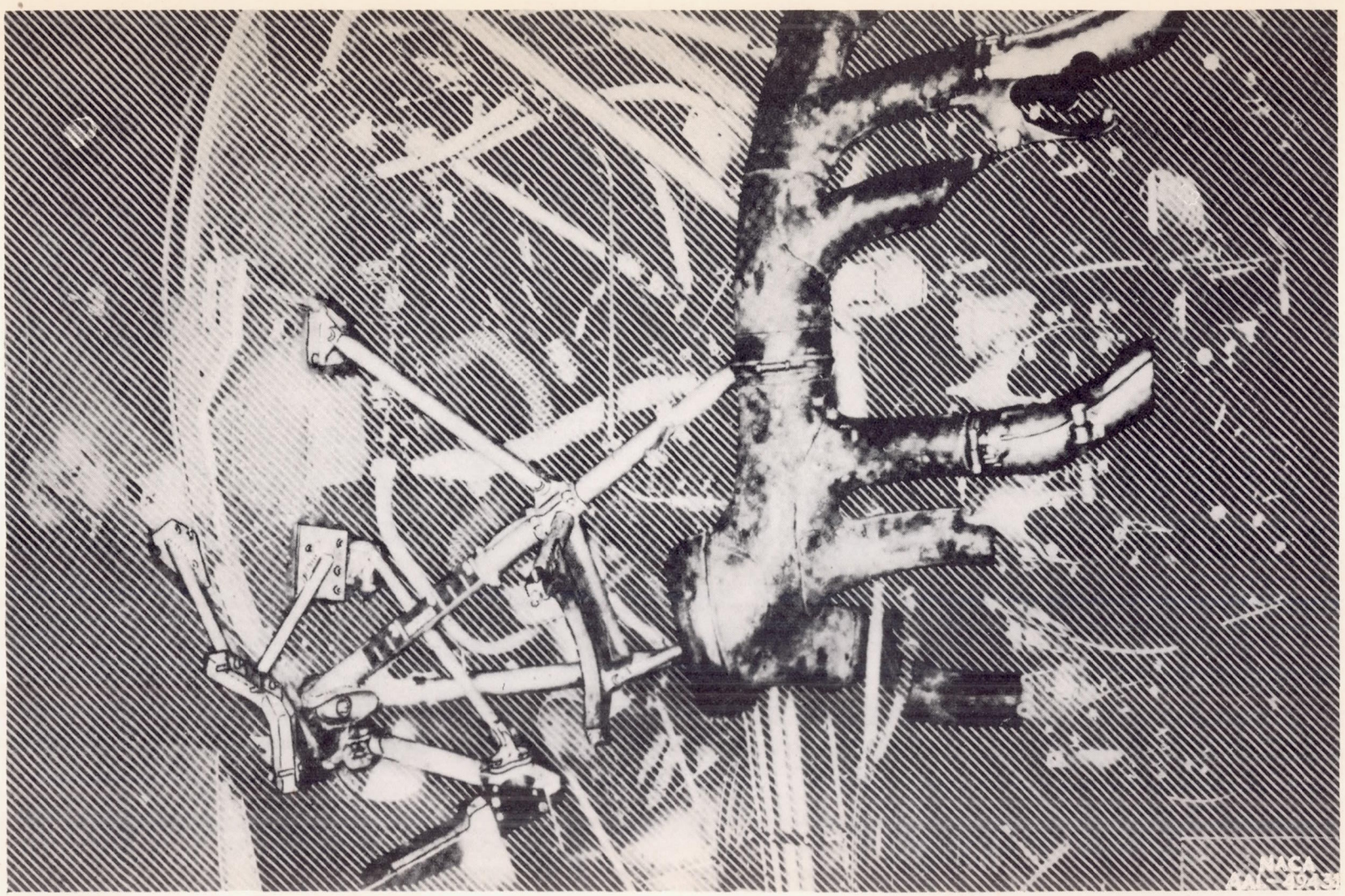

Figure 5.- Heat-exchanger supports and revised collector ring at outboard side of right nacelle, C-46 airplane.

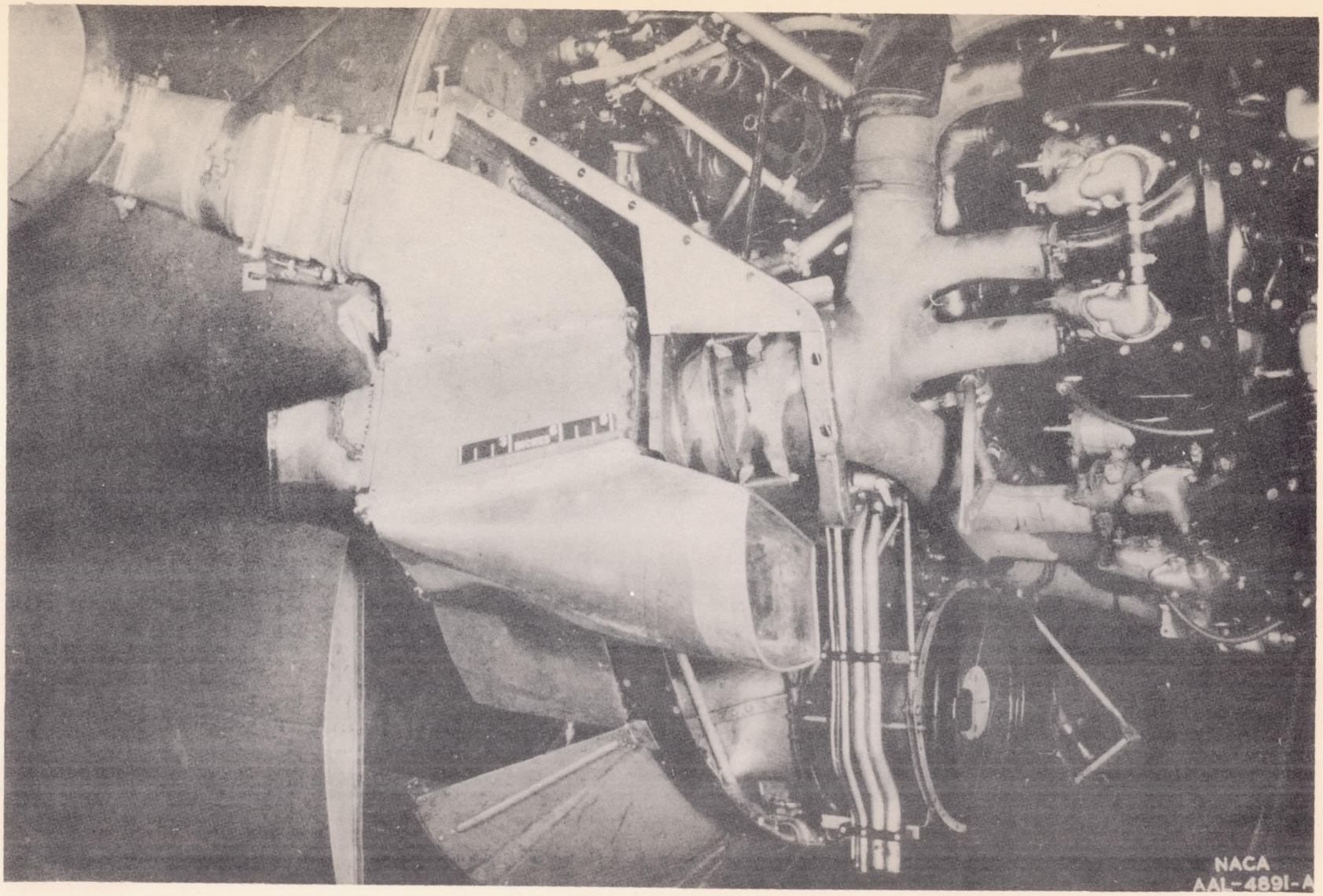

Figure 6.- Right outboard heat-exchanger installation on C-46 airplane.

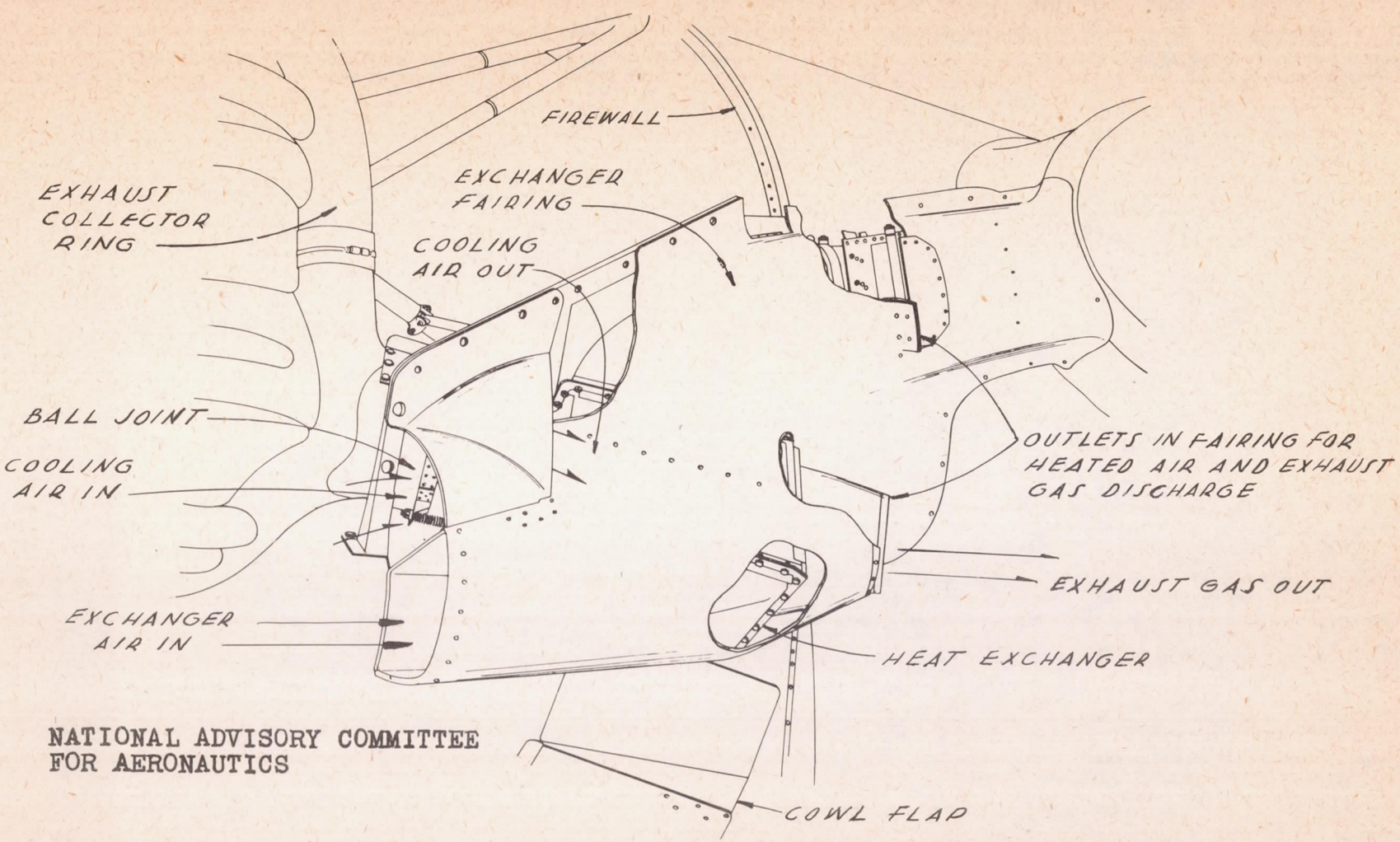

FIGURE : 7 — DETAILS OF HEAT EXCHANGER FAIRING AND BALL JOINT COOLING AIR SCOOP, C-46 AIRPLANE

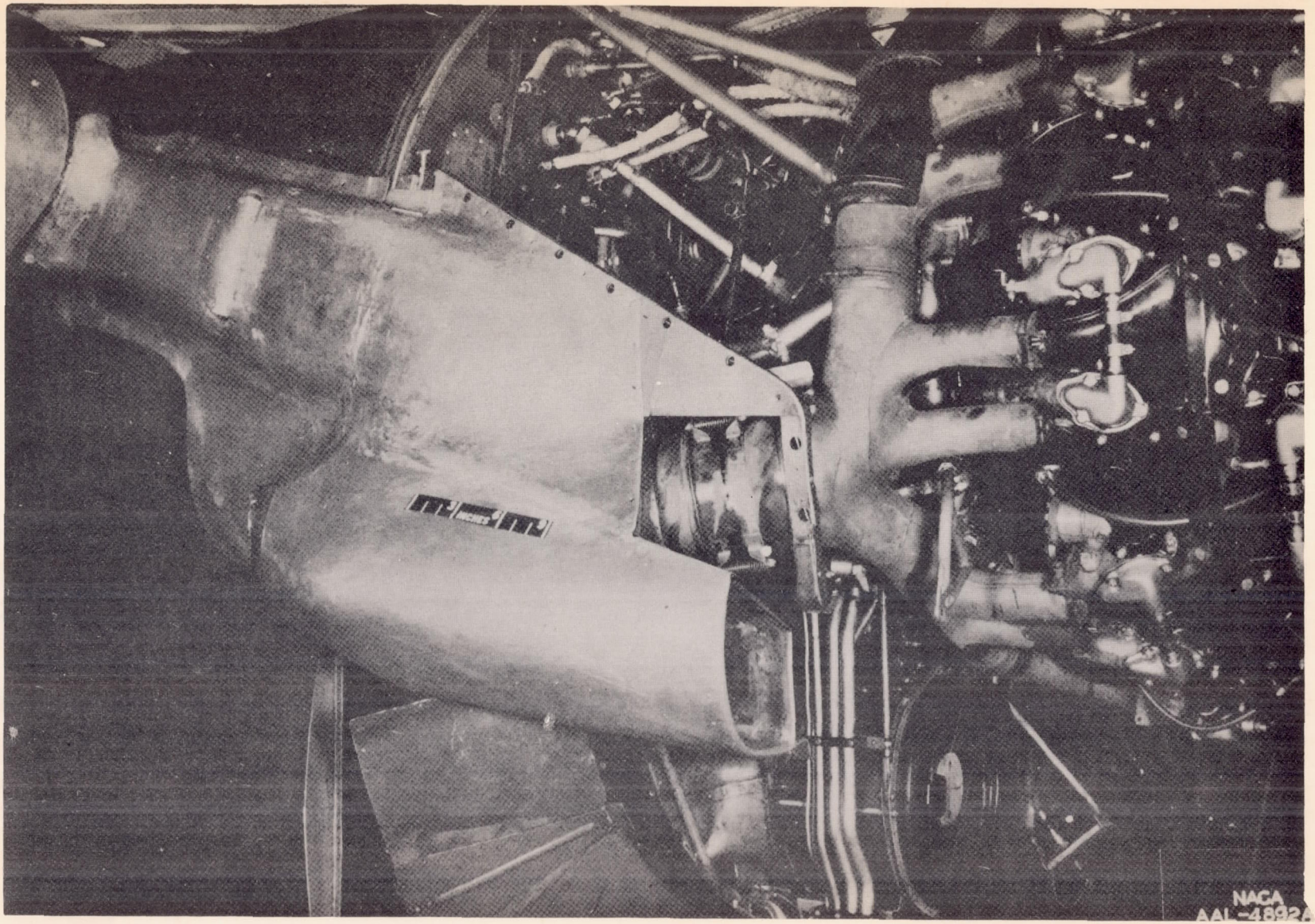

Figure 8.- Heat-exchanger shroud installed on outboard side of right nacelle, C-46 airplane.

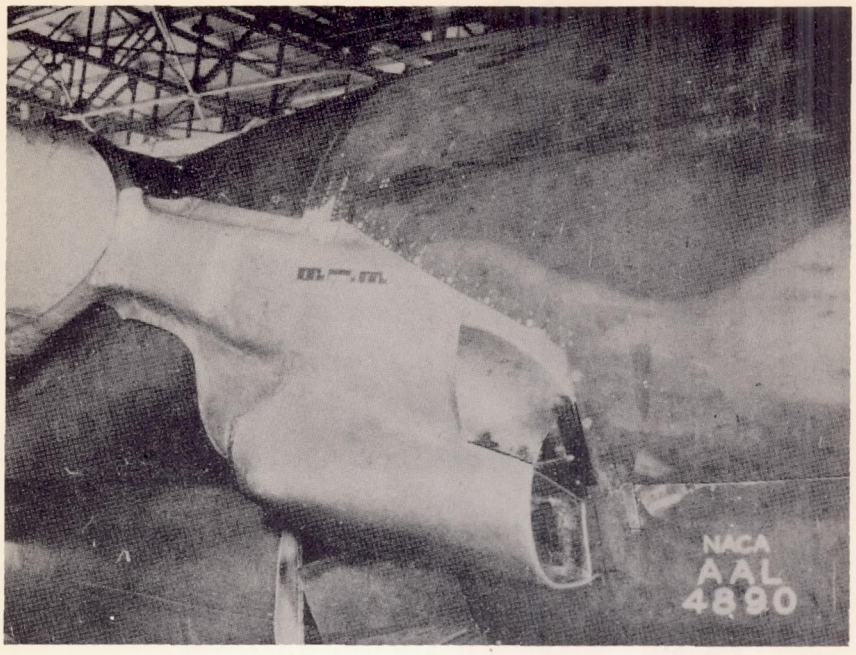

Figure 9.- Right outboard heat-exchanger installation on C-46 airplane ready for flight.

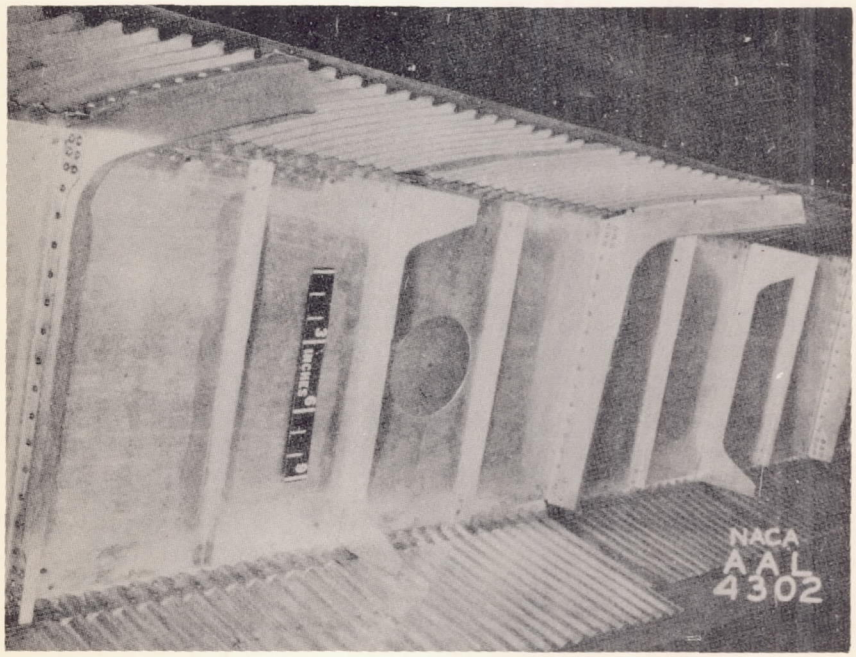

Figure 12.- Rear view of 5.5-percent-chord baffle plate installed in wing outer-panel leading edge, C-46 airplane.

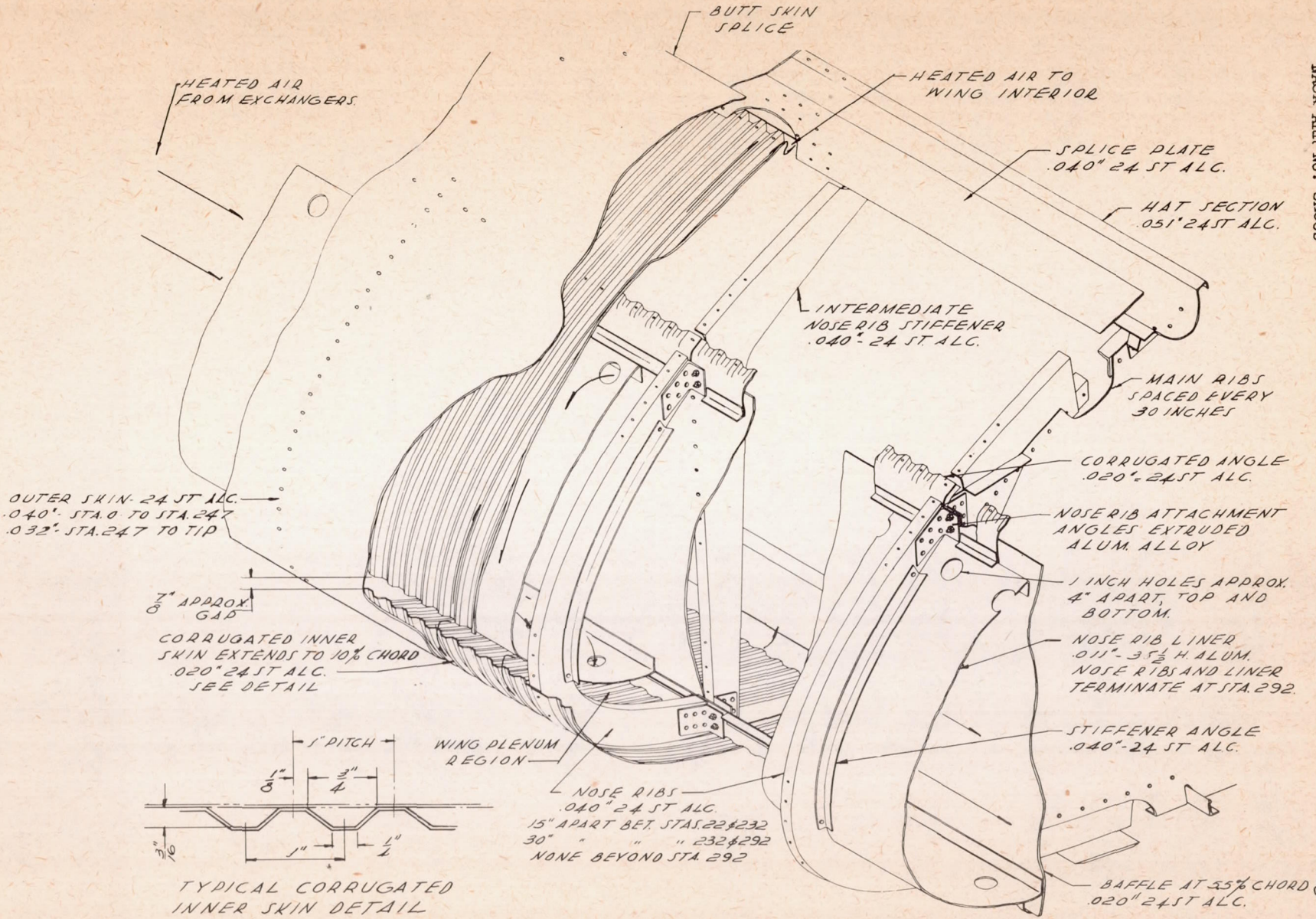

FIGURE 10— TYPICAL WING OUTER-PANEL LEADING EDGE SECTION AS REVISED FOR THERMAL ICE-PREVENTION, C-46 AIRPLANE

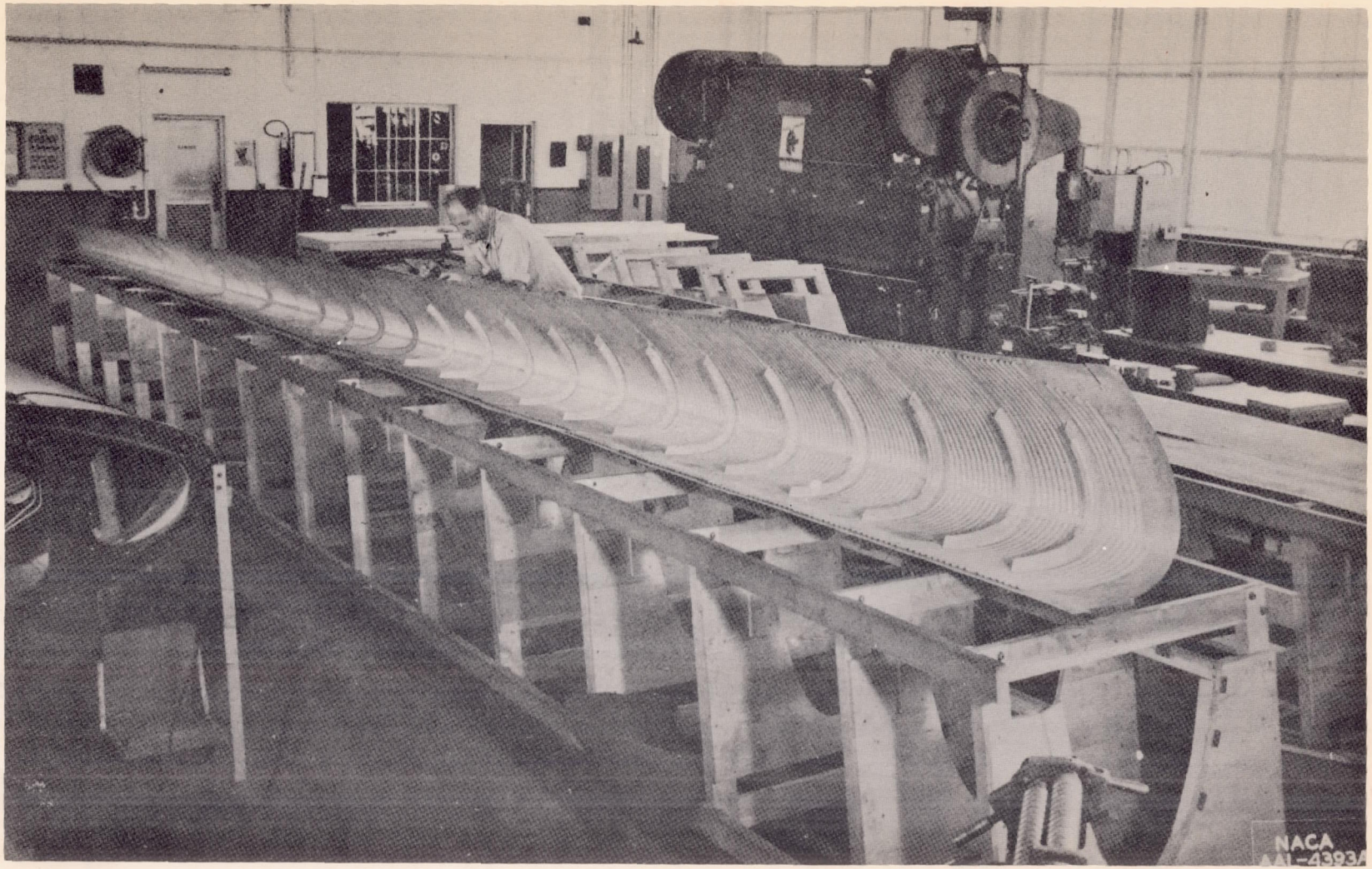

Figure 11.- Corrugated inner skin and revised nose ribs installed in leading edge of left wing outer panel, C-46 airplane.

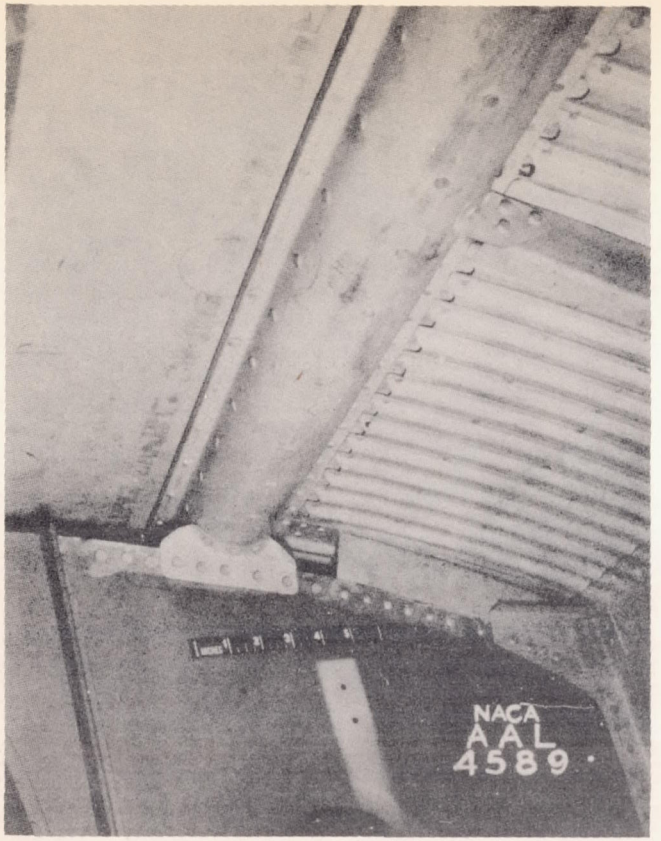

Figure 13.- View showing details of attachment of revised wing outer-panel leading edge to original structure, C-46 airplane.

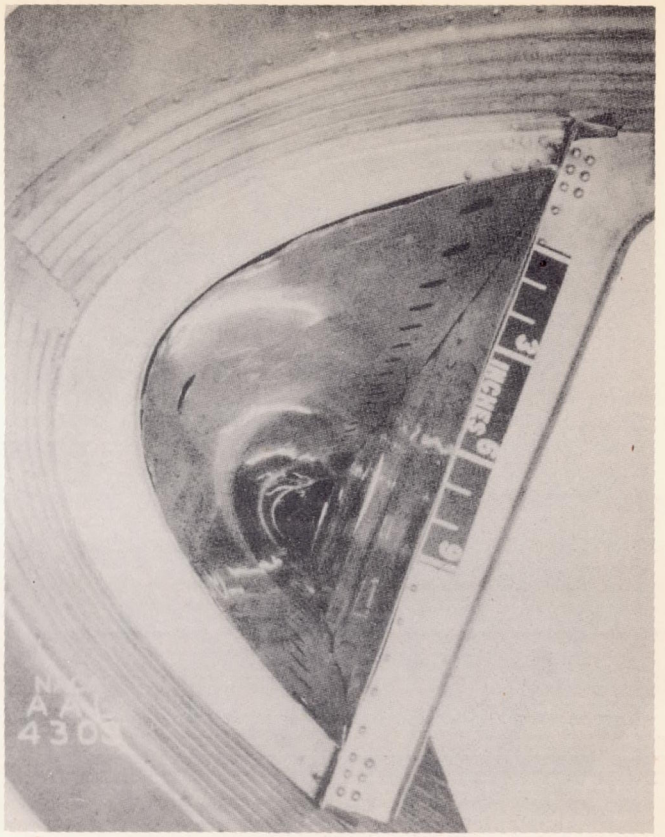

Figure 14.- View of nose liner in right wing outer-panel leading edge from inboard end, C-46 airplane.

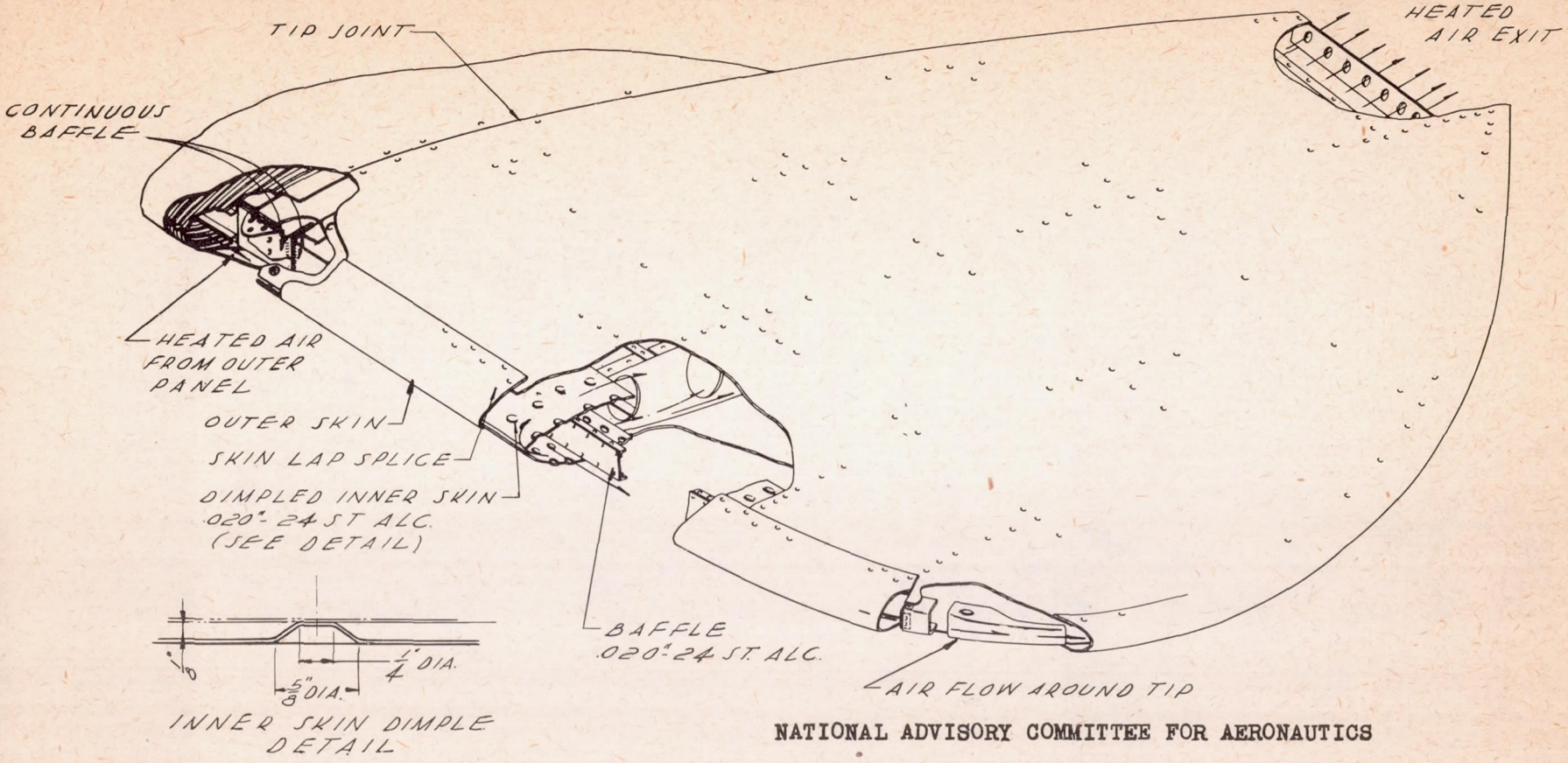

FIGURE 15 - WING TIP OF C-46 AIRPLANE AS REVISED FOR THERMAL ICE-PREVENTION

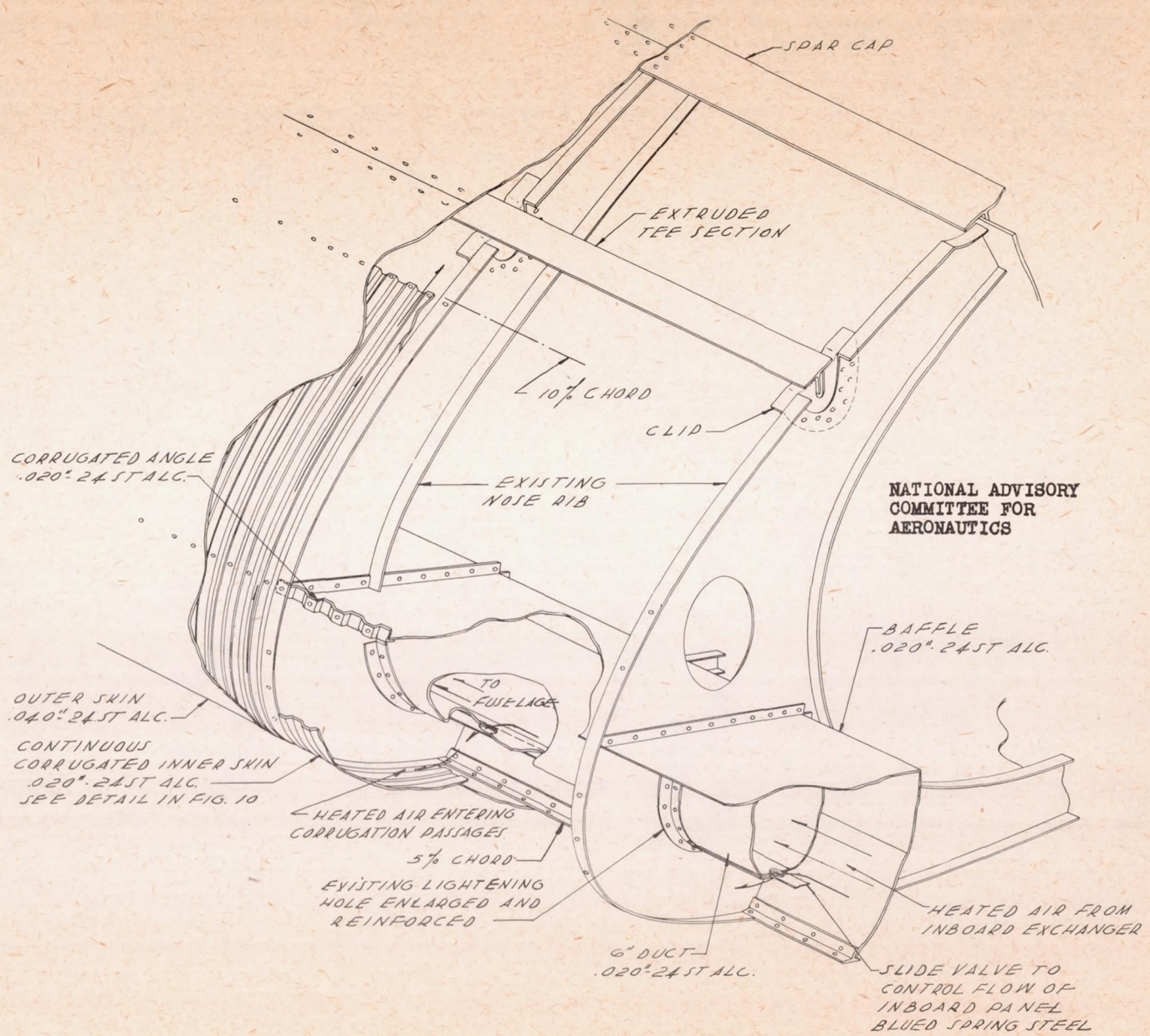

FIGURE 16 — TYPICAL WING INBOARD-PANEL LEADING-EDGE SECTION AS REVISED FOR THERMAL ICE-PREVENTION, C-46 AIRPLANE.

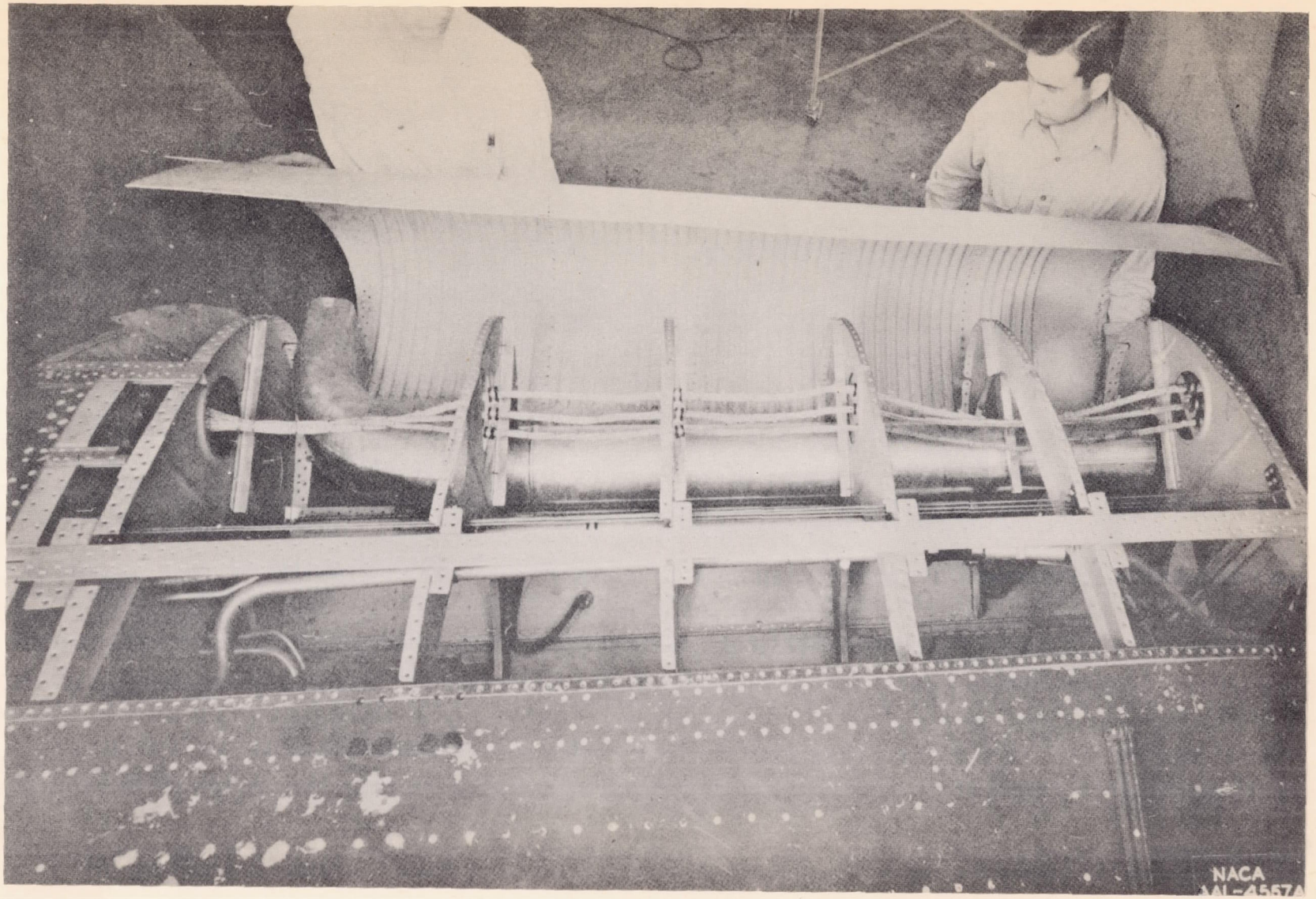

Figure 17.- Rear view of inboard-panel leading edge during installation on C-46 airplane.

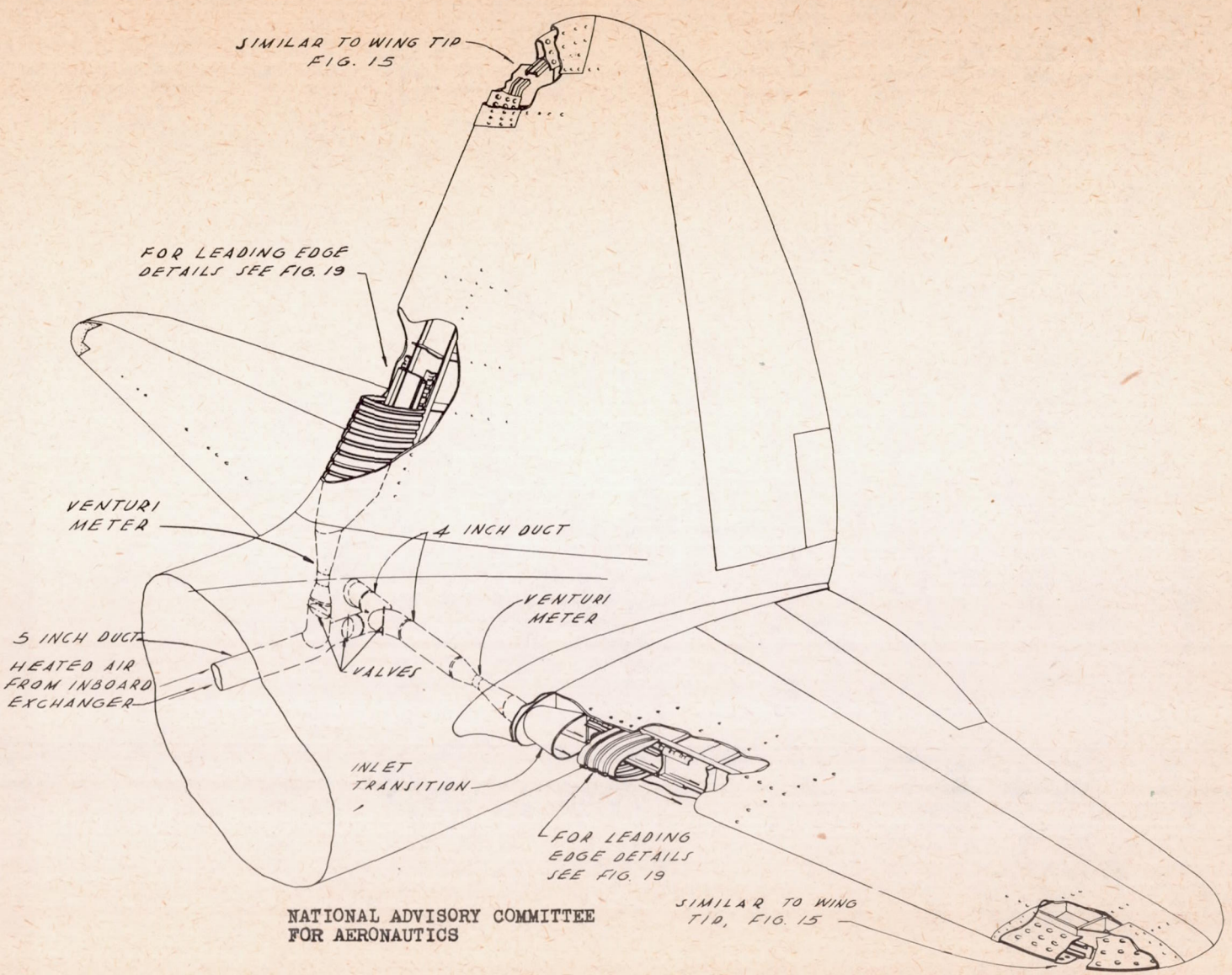

FIGURE 18.- GENERAL LAYOUT OF THERMAL ICE-PREVENTION EQUIPMENT IN EMPENNAGE OF C-46 AIRPLANE.

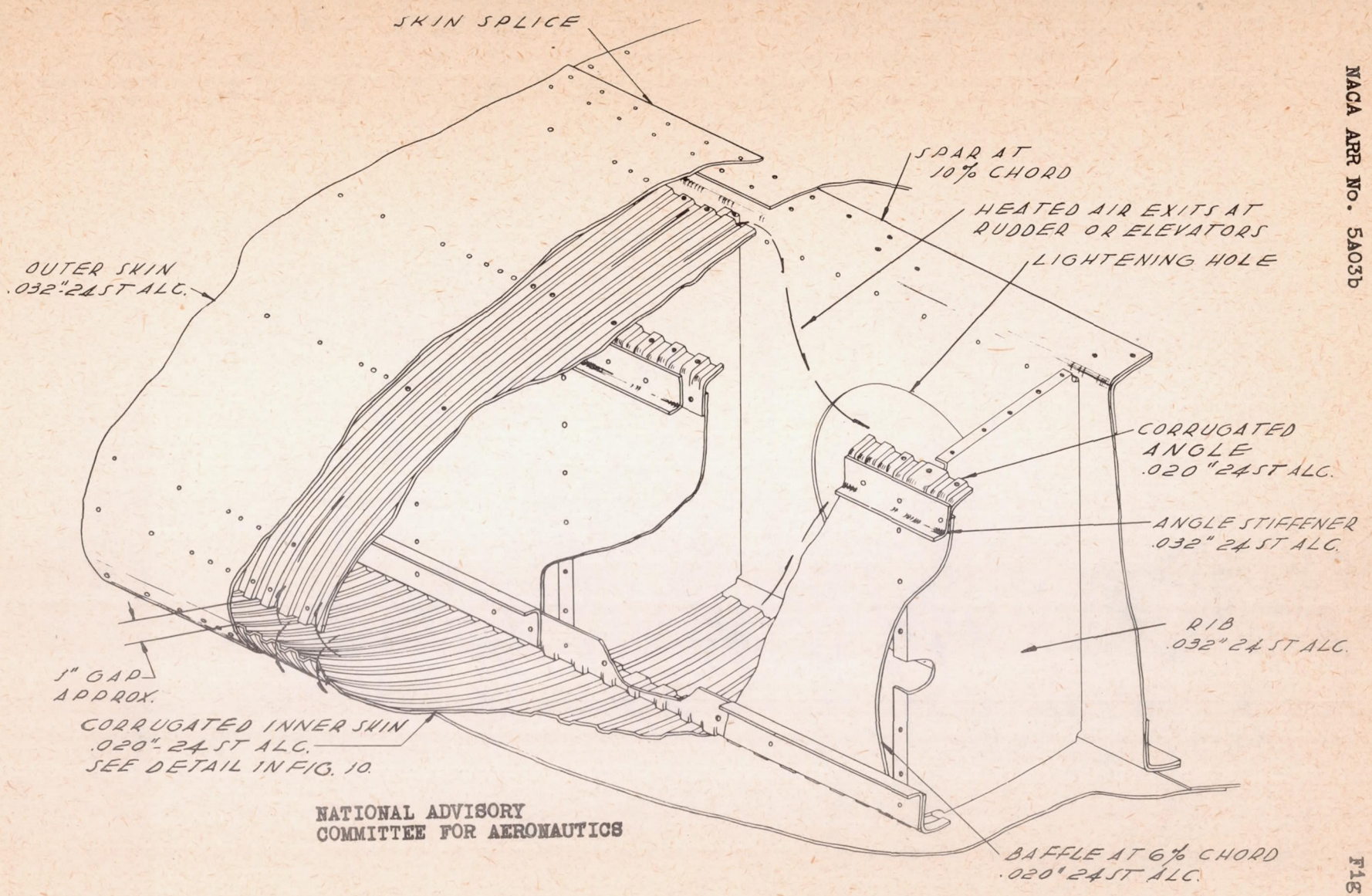

Fig. 19

FIGURE 19 DETAILS OF C-46 AIRPLANE STABILIZER AND FIN LEADING EDGES AS REVISED FOR THERMAL ICE-PREVENTION

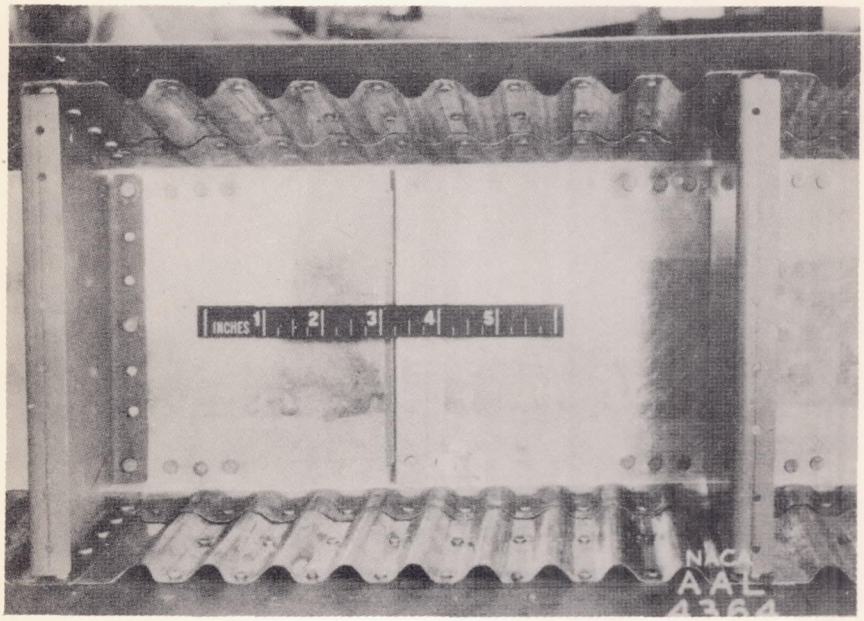

Figure 20.- Rear view of baffle plate and corrugated inner skin for stabilizer leading edge, C-46 airplane.

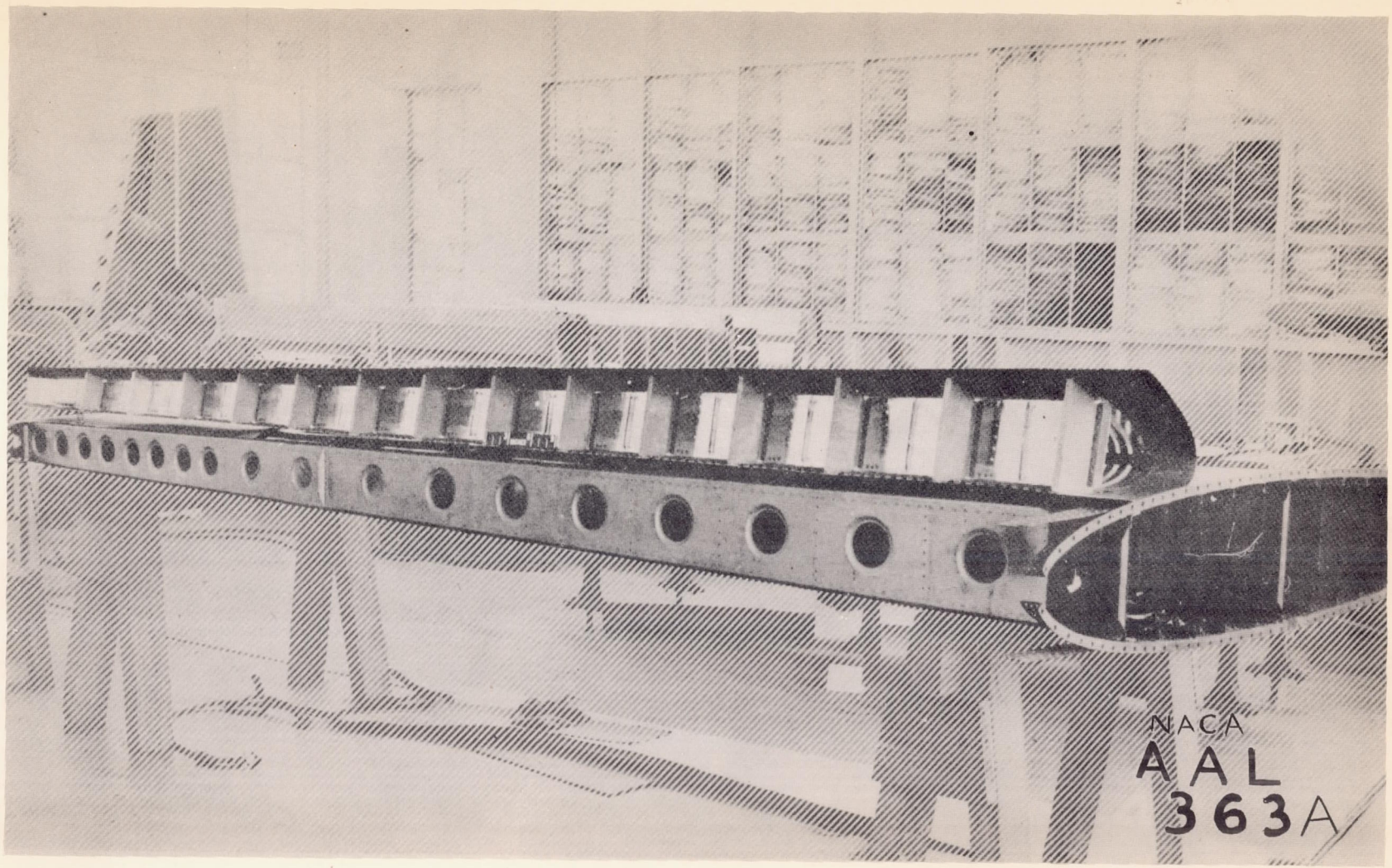

Figure 21.- Revised leading edge prior to installation on stabilizer, C-46 airplane.

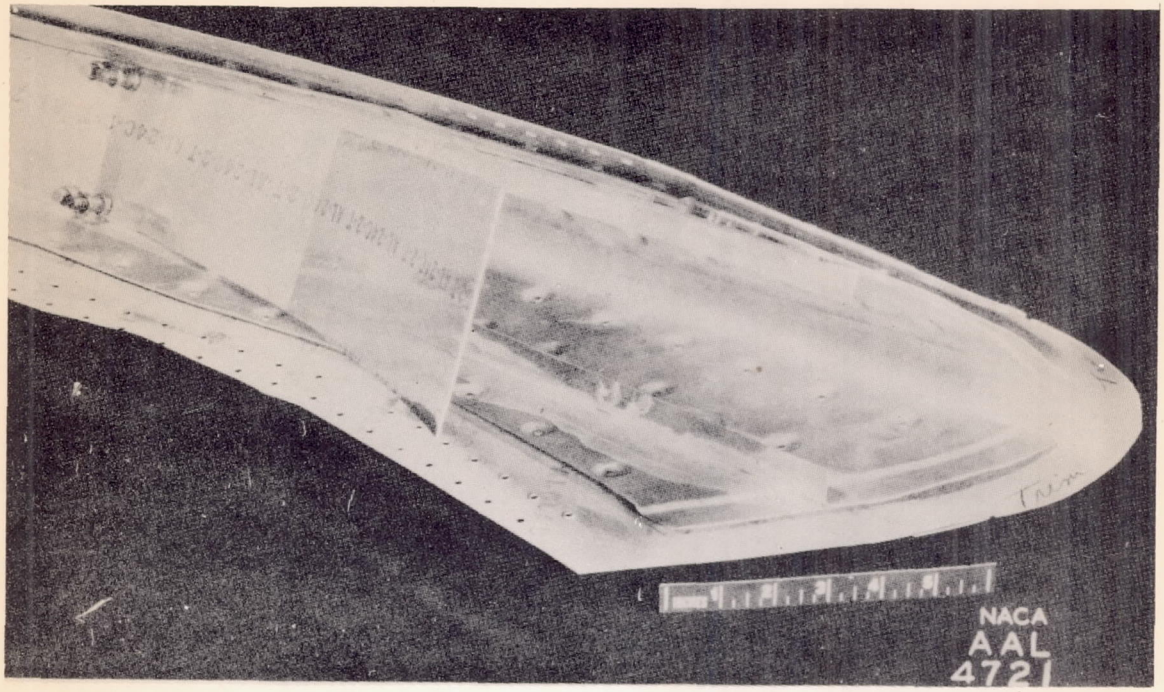

Figure 22.- Revised leading edge of fin tip showing baffle plate and dimpled-skin details, C-46 airplane.

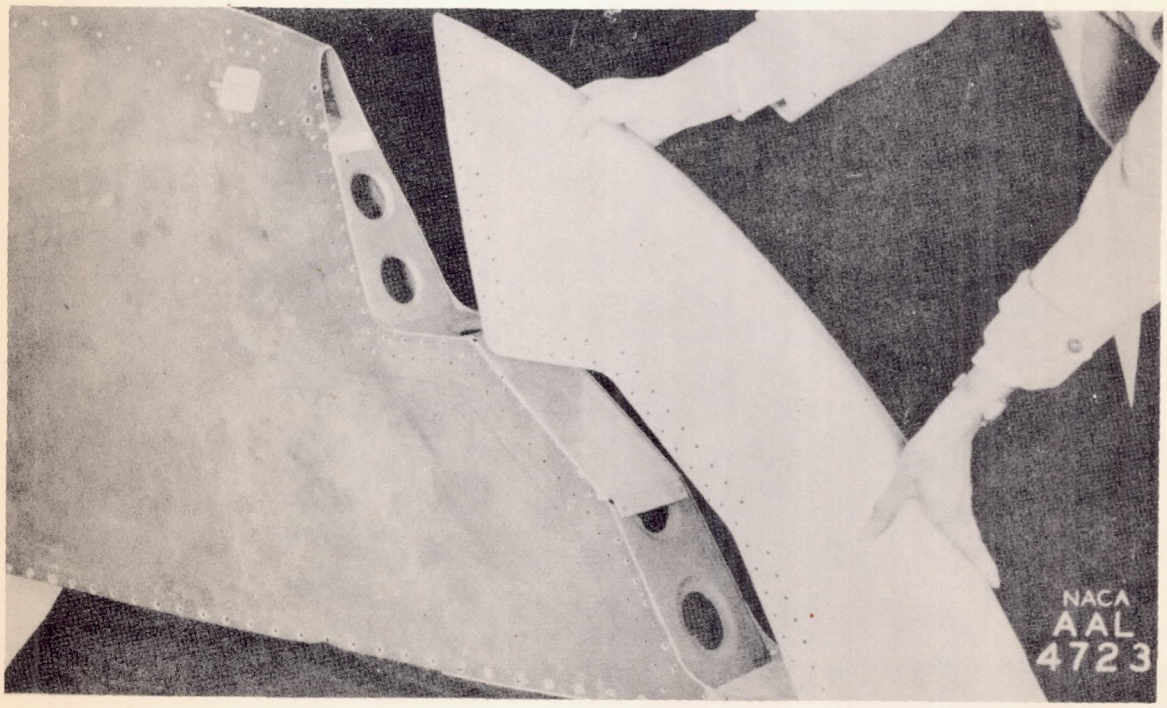

Figure 23.- Revised leading edge for fin tip prior to installation, C-46 airplane.

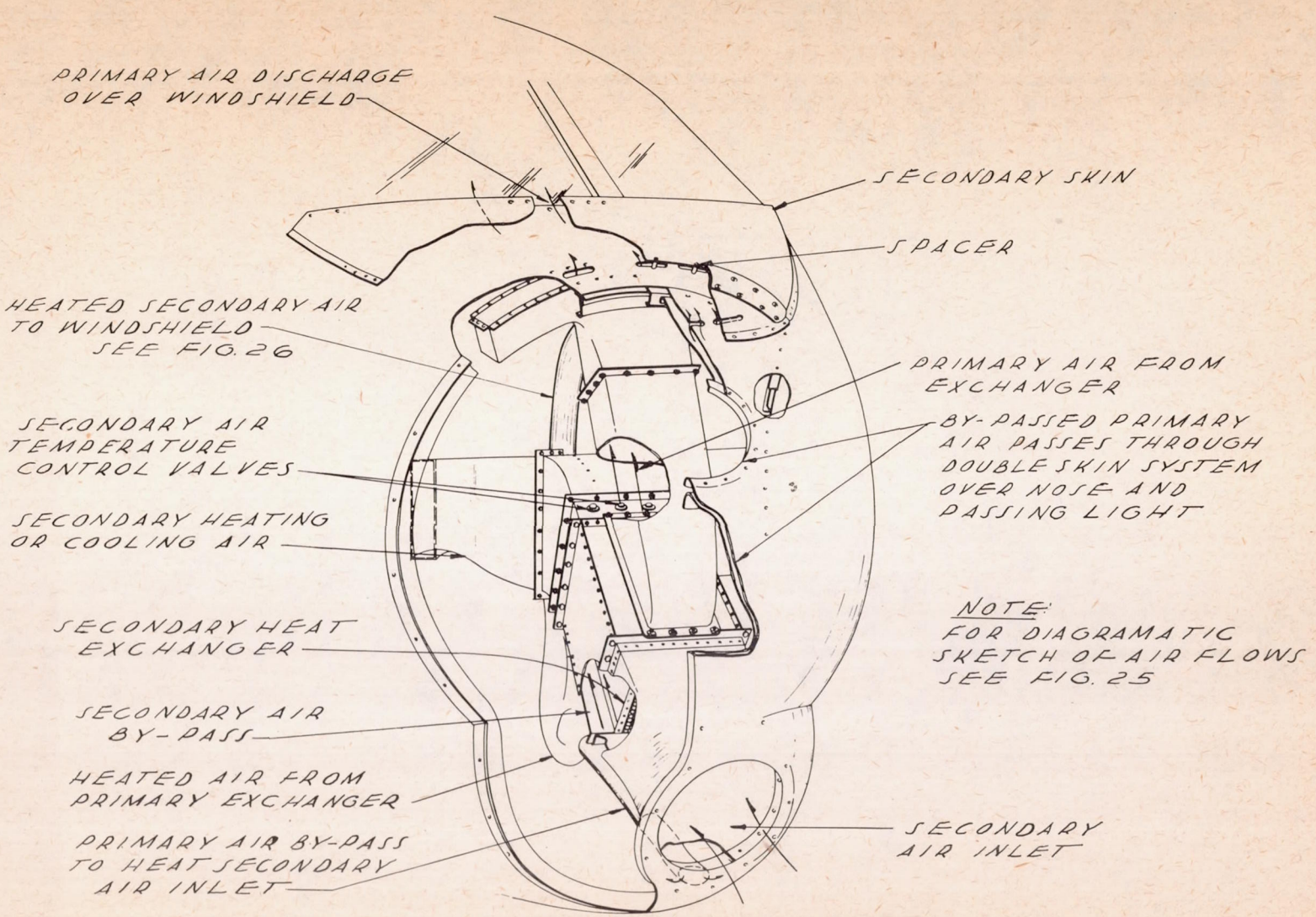

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FIGURE: 24 - DETAILS OF HEAT-EXCHANGER INSTALLED IN NOSE OF C-46 AIRPLANE TO PROVIDE SECONDARY HEATED AIR FOR WINDSHIELD AND COCKPIT HEATING.

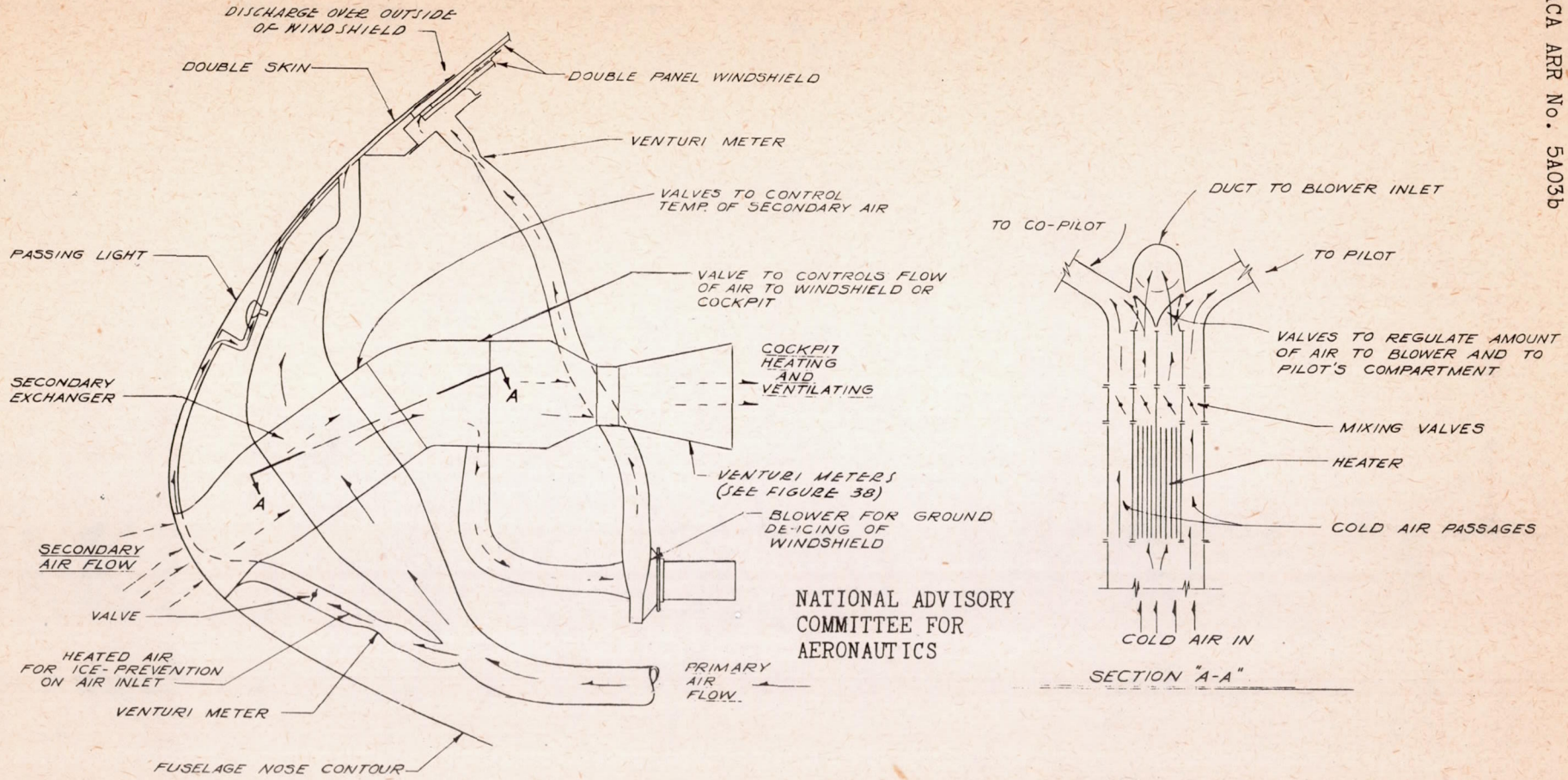


FIGURE 25.— PRIMARY AND SECONDARY AIR FLOW FOR SECONDARY HEAT-EXCHANGER INSTALLATION IN C-46 AIRPLANE.

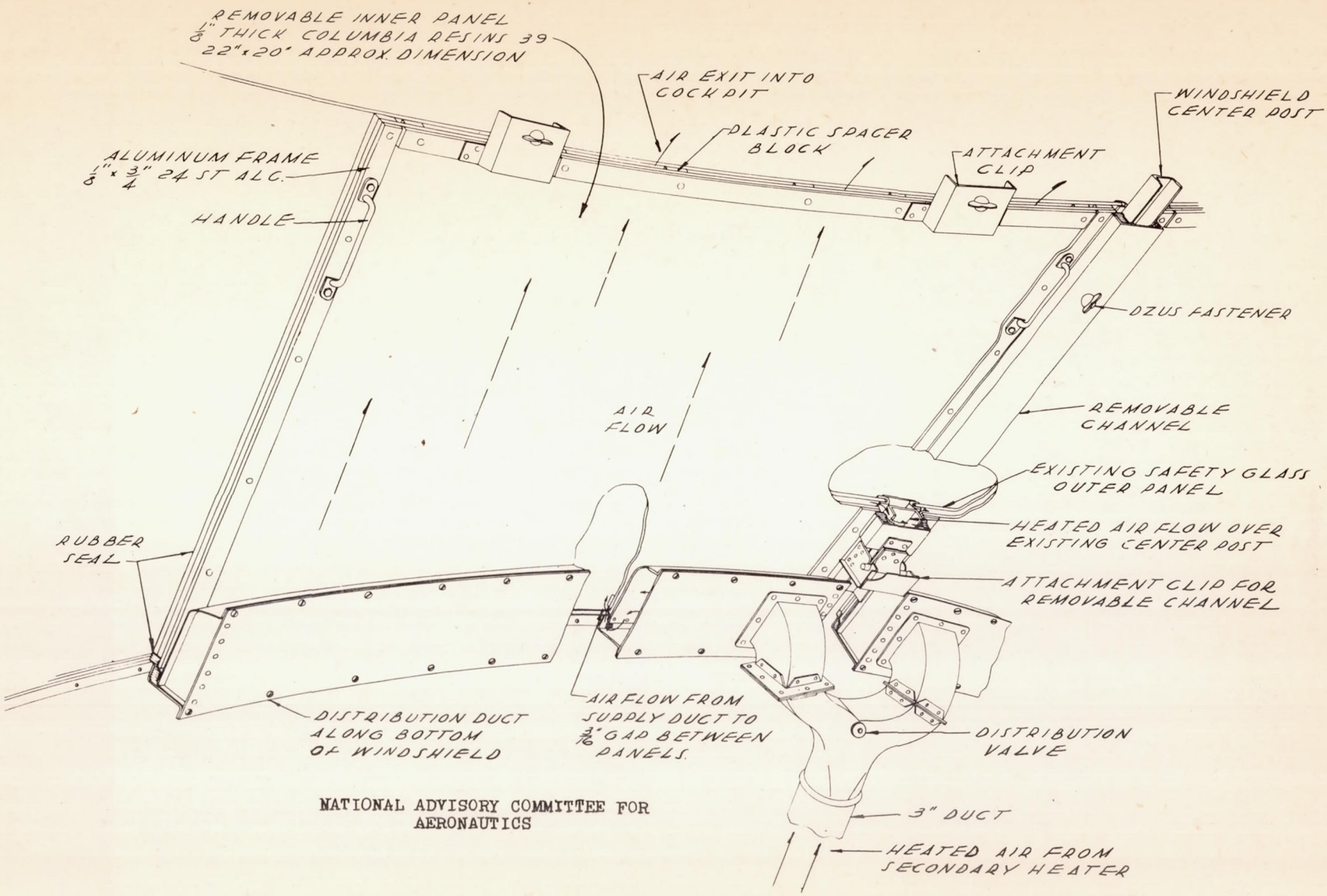


FIGURE 26 INSIDE VIEW OF PILOT'S HEATED WINDSHIELD IN C-46 AIRPLANE CO-PILOT'S WINDSHIELD IS PROVIDED WITH SIMILAR INSTALLATION

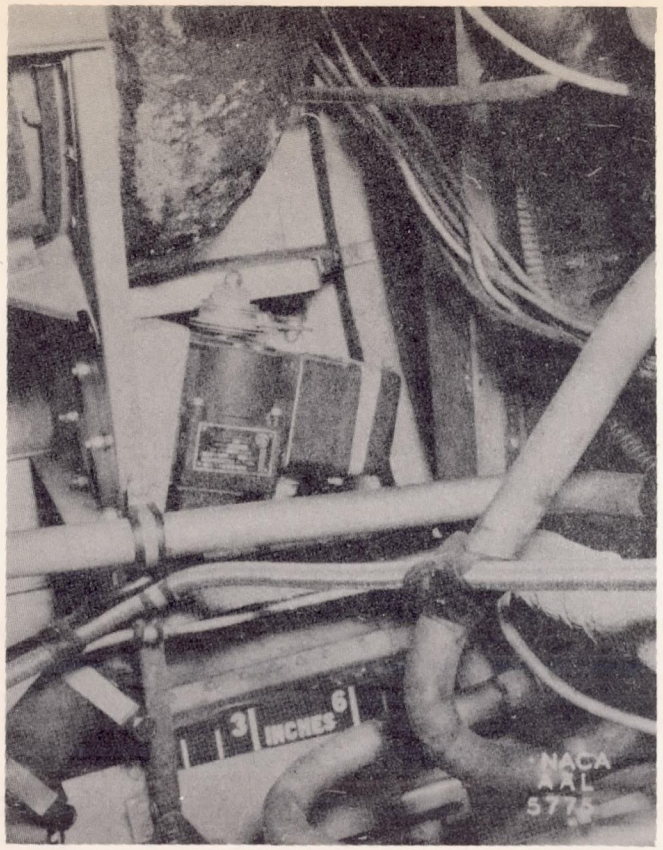


Figure 27.- Discharge-valve control motor installed in outboard side of left nacelle, C-46 airplane.

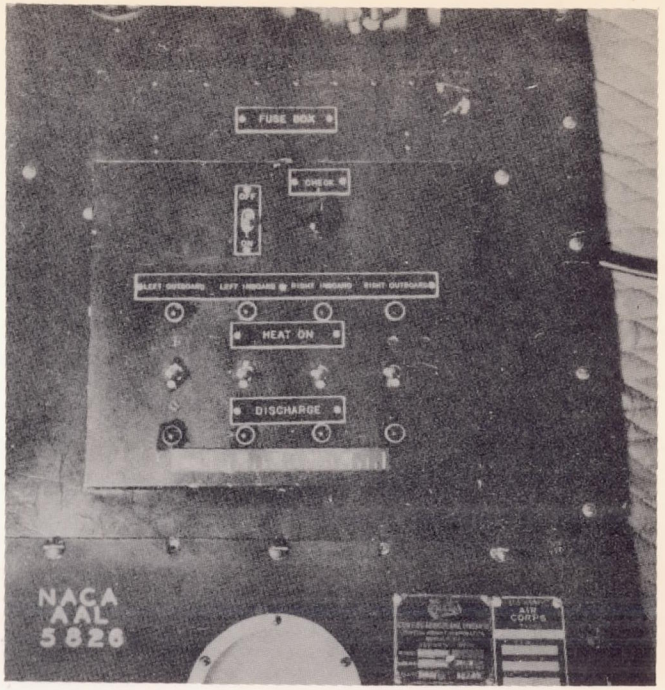
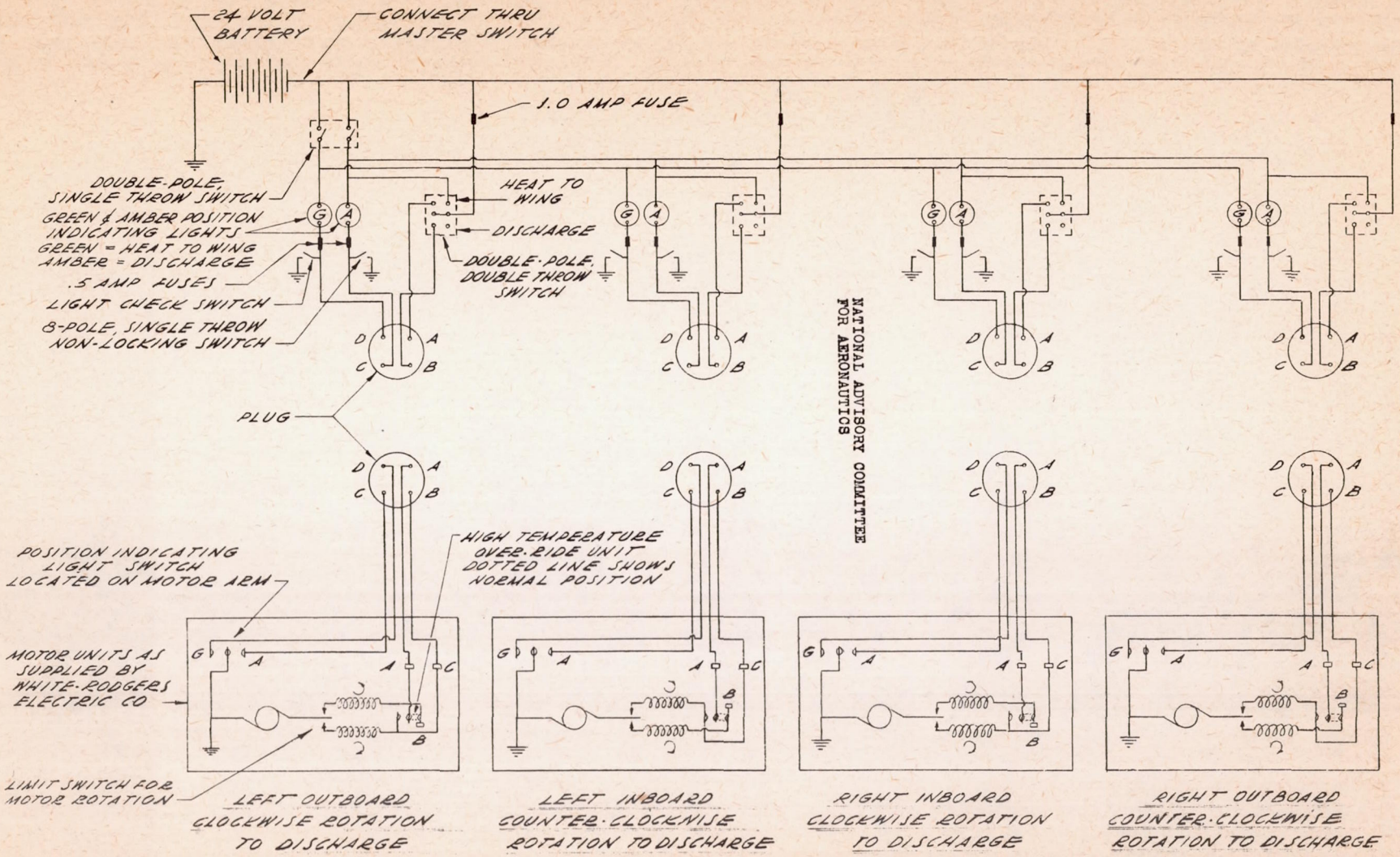


Figure 29.- Control panel for discharge-valve motors of C-46 airplane thermal ice-prevention equipment.



NOTES: 1- WIRING TO BE IDENTICAL FOR ALL FOUR MOTORS EXCEPT AS NOTED
2-FOR INSTRUMENT PANEL SEE FIGURE 29

FIGURE 28.- WIRING DIAGRAM OF CONTROL MOTORS FOR DISCHARGE VALVES LOCATED AT HEAT-EXCHANGER AIR OUTLETS, C-46 AIRPLANE.

Fig. 28

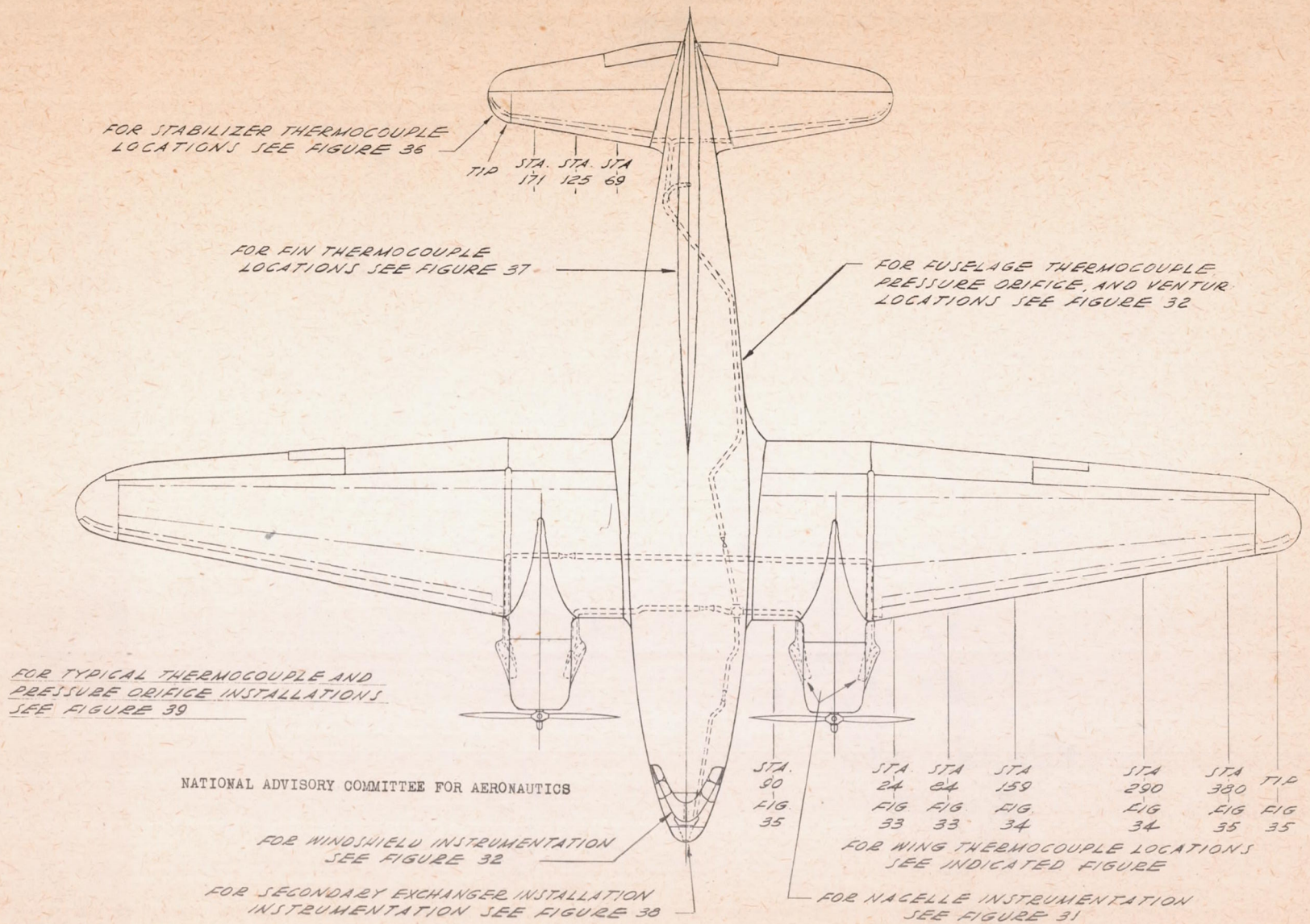
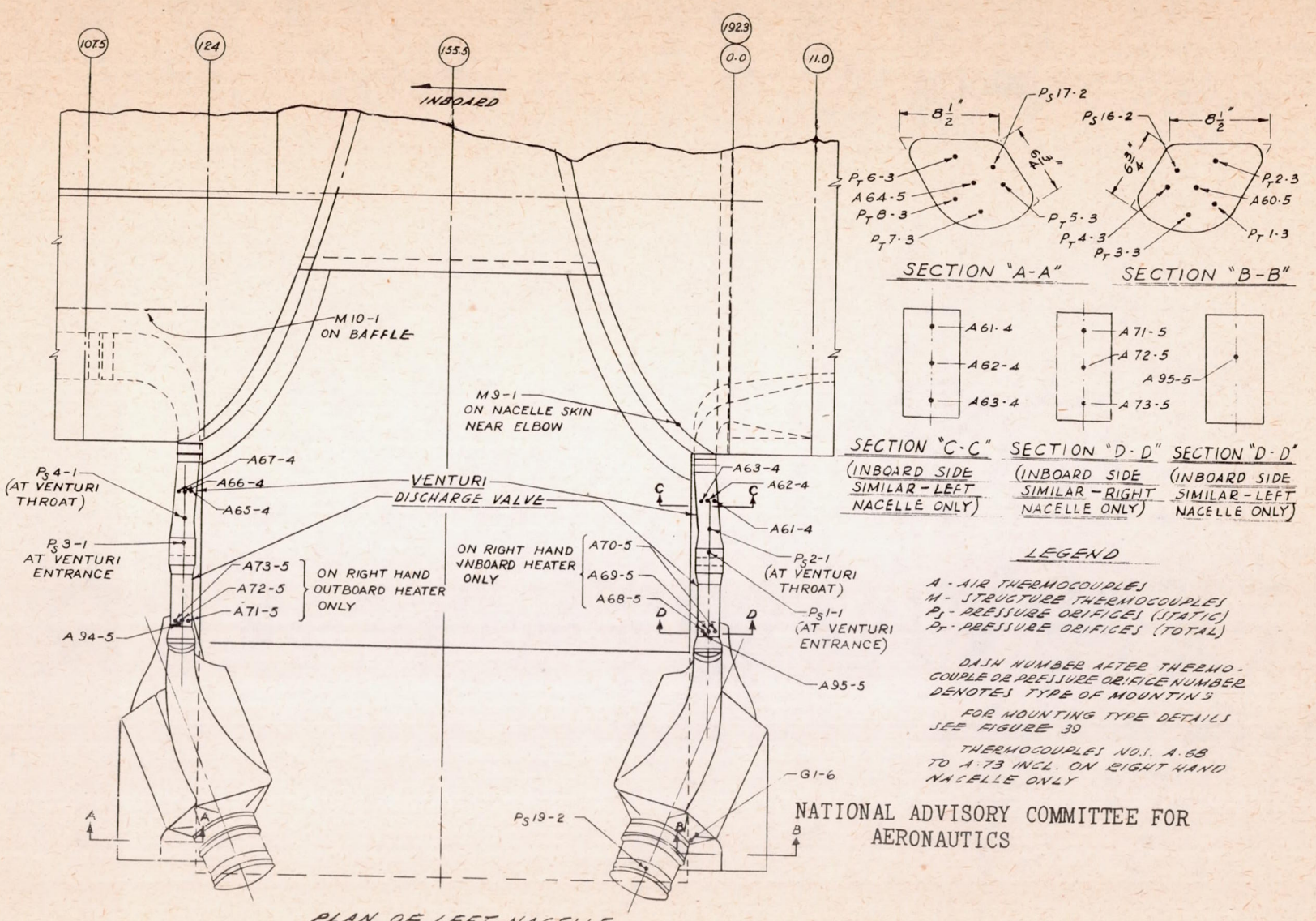
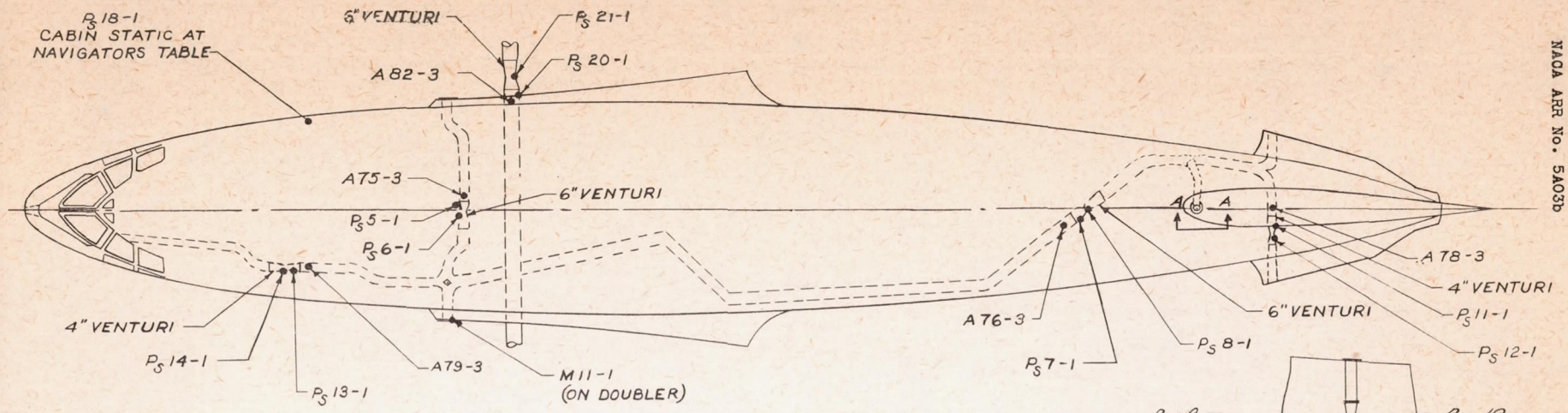


FIGURE 30. - INDEX TO THERMOCOUPLE AND PRESSURE-ORIFICE LOCATIONS ON C-46 AIRPLANE.



PLAN OF LEFT NACELLE

FIGURE 31. - THERMOCOUPLE AND PRESSURE-ORIFICE LOCATIONS FOR HEAT-EXCHANGER INSTALLATION IN LEFT NACELLE OF C-46 AIRPLANE.



PLAN OF FUSELAGE

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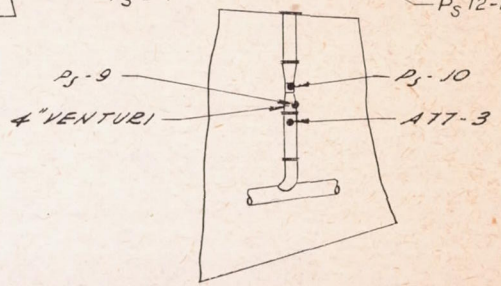
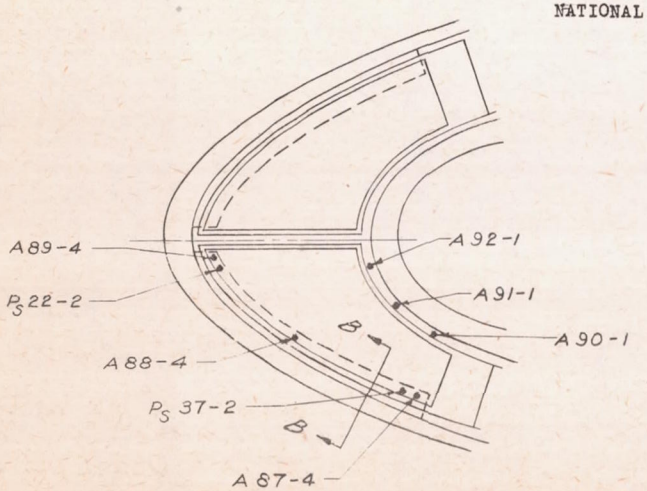
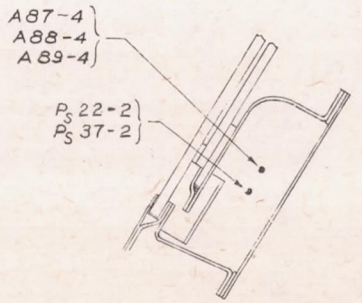


VIEW "A-A"
SHOWING DUCT TO FIN



PLAN OF WINDSHIELD



SECTION "B-B"
WINDSHIELD SUPPLY DUCT

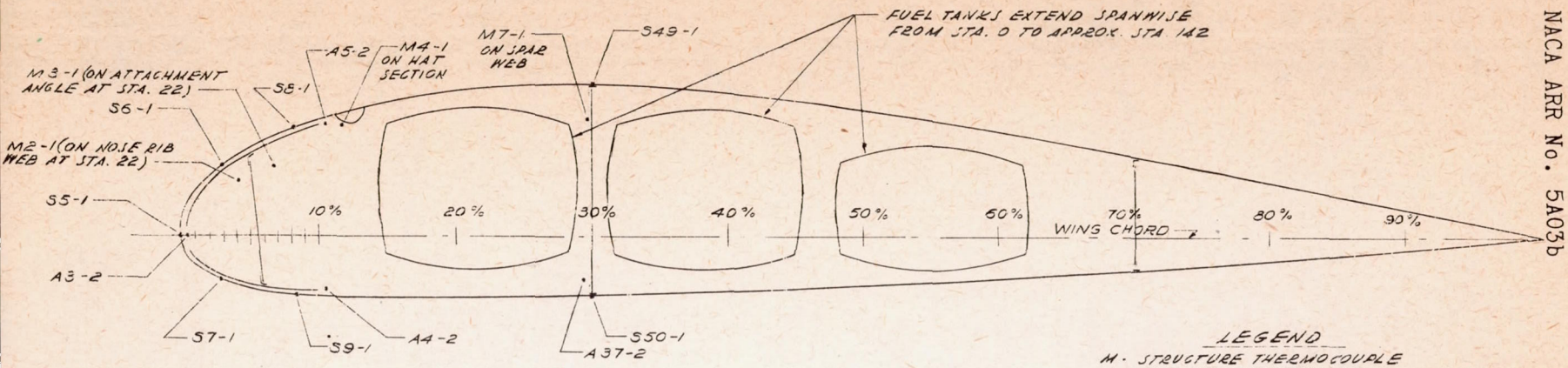
LEGEND

A - AIR THERMOCOUPLES
P_S - PRESSURE ORIFICES (STATIC)
M - STRUCTURE THERMOCOUPLES

DASH NUMBER AFTER THERMOCOUPLE OR PRESSURE ORIFICE NUMBER DENOTES TYPE OF MOUNTING FOR MOUNTING TYPE DETAILS SEE FIGURE 39

FIGURE 32.- THERMOCOUPLE AND PRESSURE-ORIFICE LOCATIONS FOR FUSELAGE DUCTS AND WINDSHIELD OF C-46 AIRPLANE.

NACA ARR No. 5A03b



STATION 24

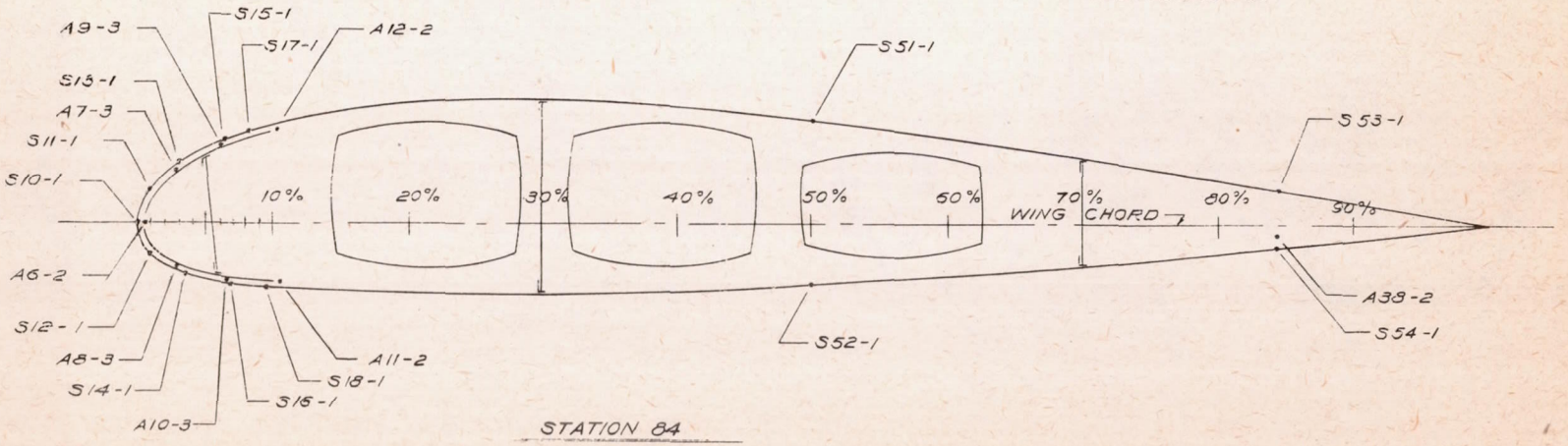
LEGEND

M - STRUCTURE THERMOCOUPLE
A - AIR THERMOCOUPLES
S - SKIN THERMOCOUPLES

DASH NUMBER AFTER THERMOCOUPLE NUMBER DENOTES TYPE OF MOUNTING

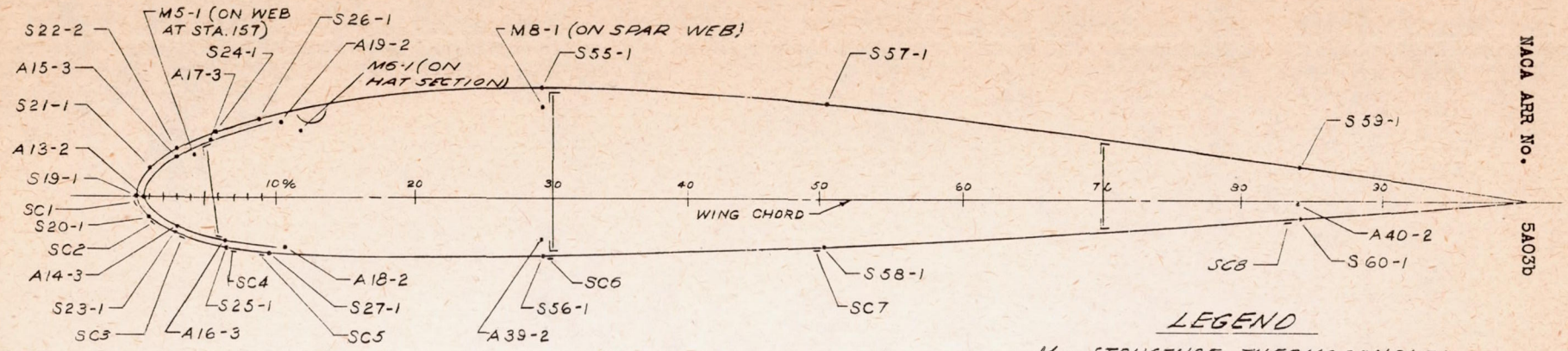
FOR MOUNTING TYPE DETAILS SEE FIGURE 39

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS



STATION 84

FIGURE 33.- THERMOCOUPLE LOCATIONS FOR WING OUTER PANEL - STATIONS 24 AND 84 - C-46 AIRPLANE.



STATION 159

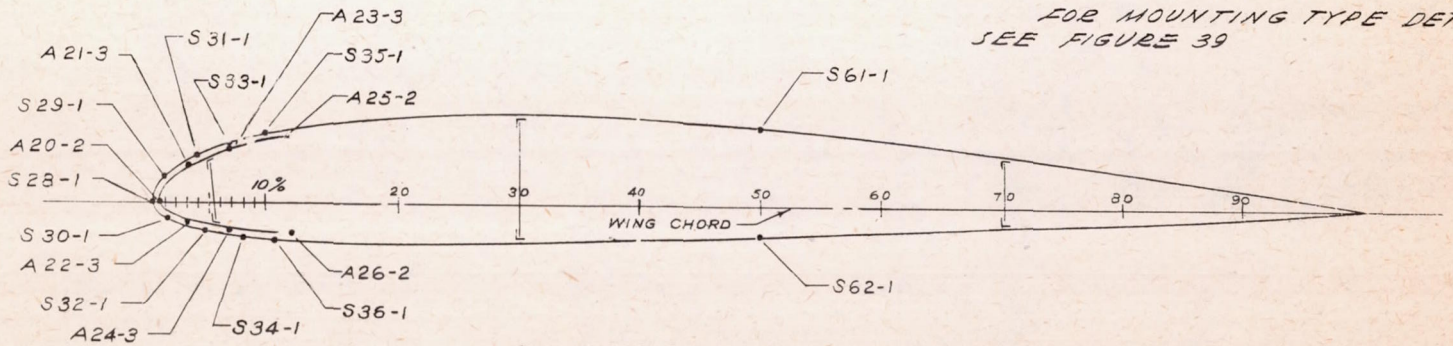
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

LEGEND

M - STRUCTURE THERMOCOUPLES
A - AIR THERMOCOUPLES
S - SKIN THERMOCOUPLES
SC - SURFACE THERMOCOUPLES

DASH NUMBER AFTER THERMOCOUPLE NUMBER DENOTES TYPE OF MOUNTING

FOR MOUNTING TYPE DETAILS SEE FIGURE 39

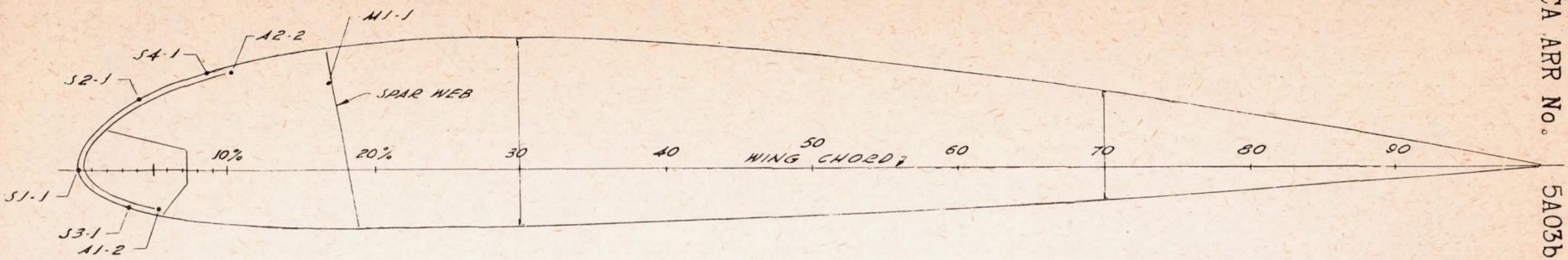


STATION 290

FIGURE 34.- THERMOCOUPLE LOCATIONS FOR WING OUTER PANEL - STATIONS 159 AND 290 - C-46 AIRPLANE

NACA ARR No. 5A03b

Fig. 34



STATION 90

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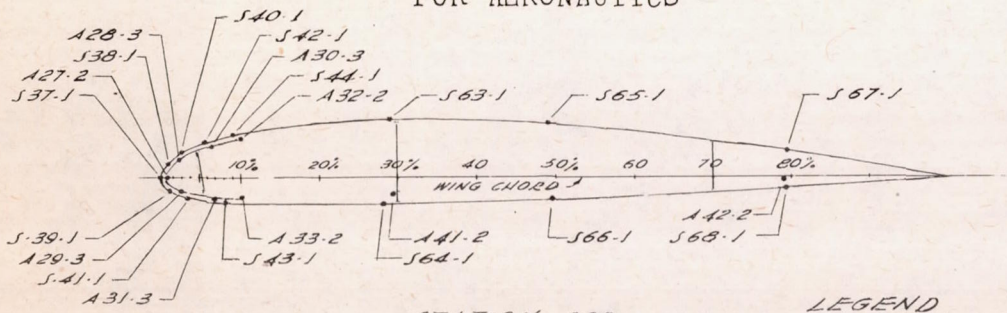
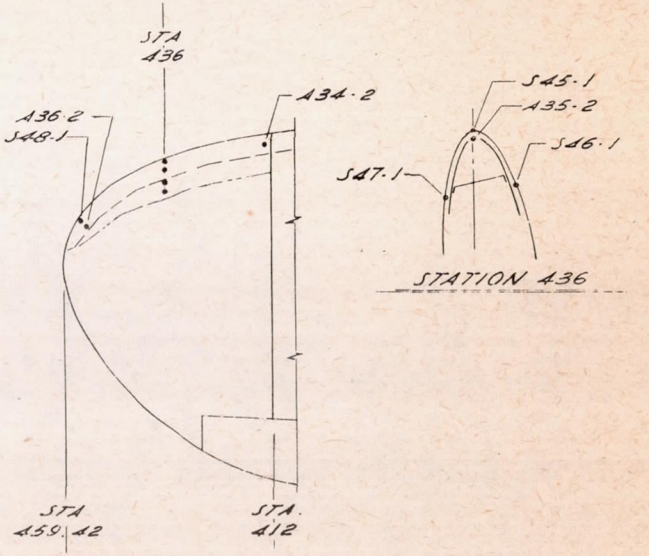


STATION 380



WING TIP

LEGEND

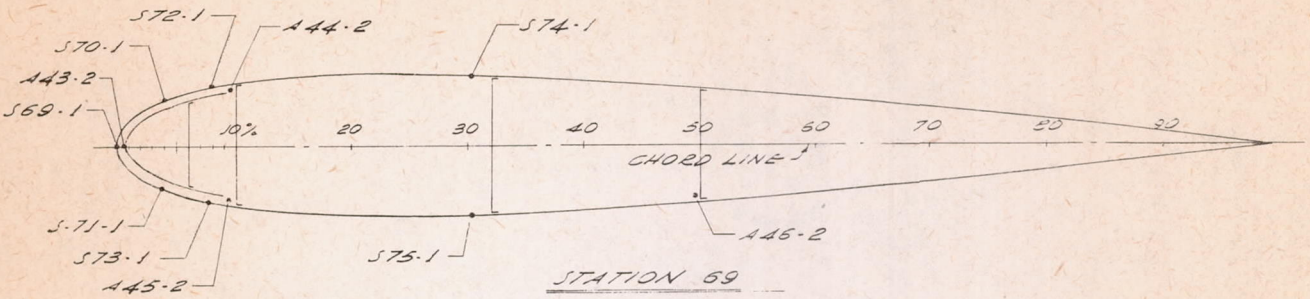
A - AIR THERMOCOUPLES
S - SKIN THERMOCOUPLES
M - STRUCTURE THERMOCOUPLES

DASH NUMBER AFTER THERMOCOUPLE NUMBER DENOTES TYPE OF MOUNTING

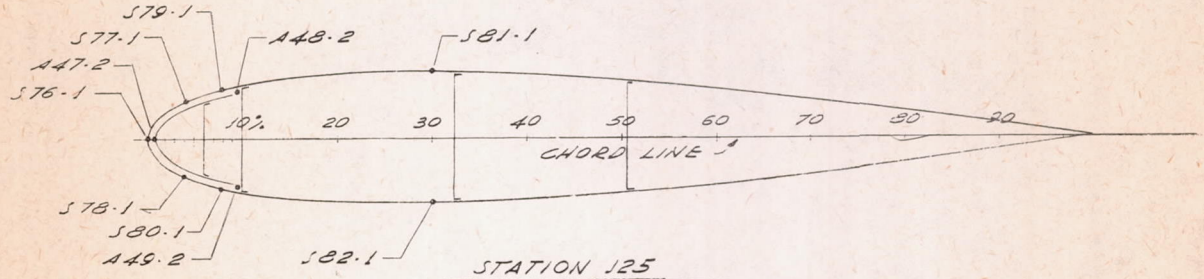
FOR MOUNTING TYPE DETAILS SEE FIG. 39

FIGURE 35 - THERMOCOUPLE LOCATIONS FOR WING OUTER PANEL STATION 380, WING TIP AND INBOARD PANEL STATION 90 - G46 AIRPLANE

A-16



STATION 69



STATION 125

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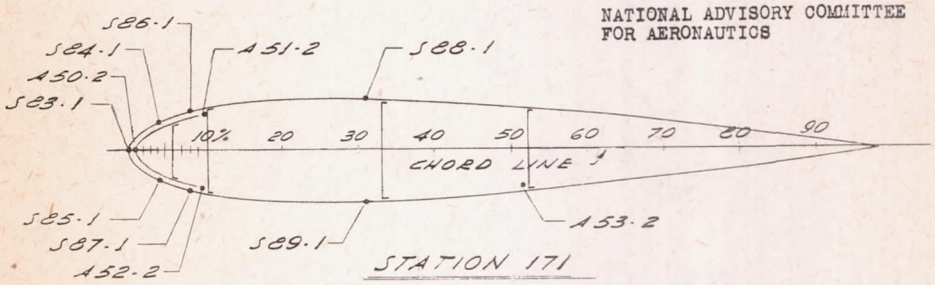
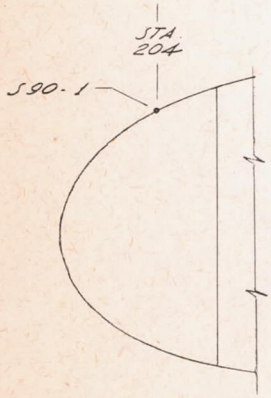
FOR AERONAUTICS



STATION 171



TIP

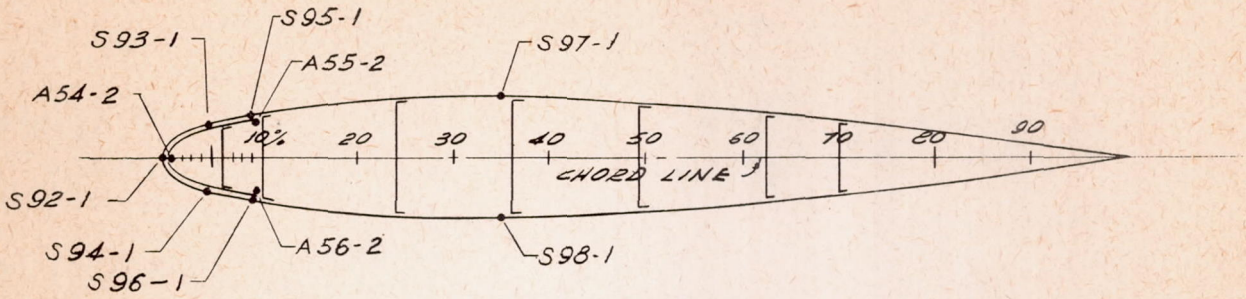
LEGEND

A - AIR THERMOCOUPLES
S - SKIN THERMOCOUPLES

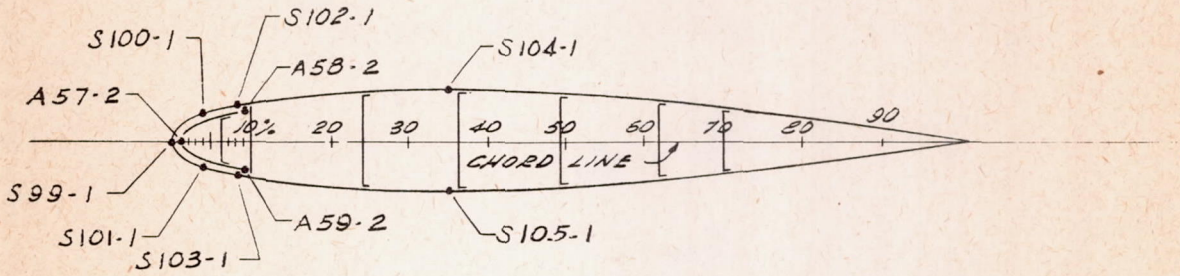
DASH NUMBER AFTER THERMOCOUPLE NUMBER DENOTES TYPE OF MOUNTING

FOR MOUNTING TYPE DETAILS SEE FIGURE 39

FIGURE 36 - THERMOCOUPLE LOCATIONS FOR STABILIZER - C-46 AIRPLANE



STATION 124



STATION 170

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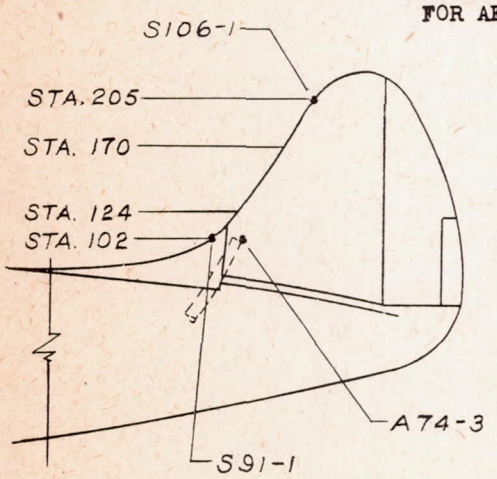


FIN

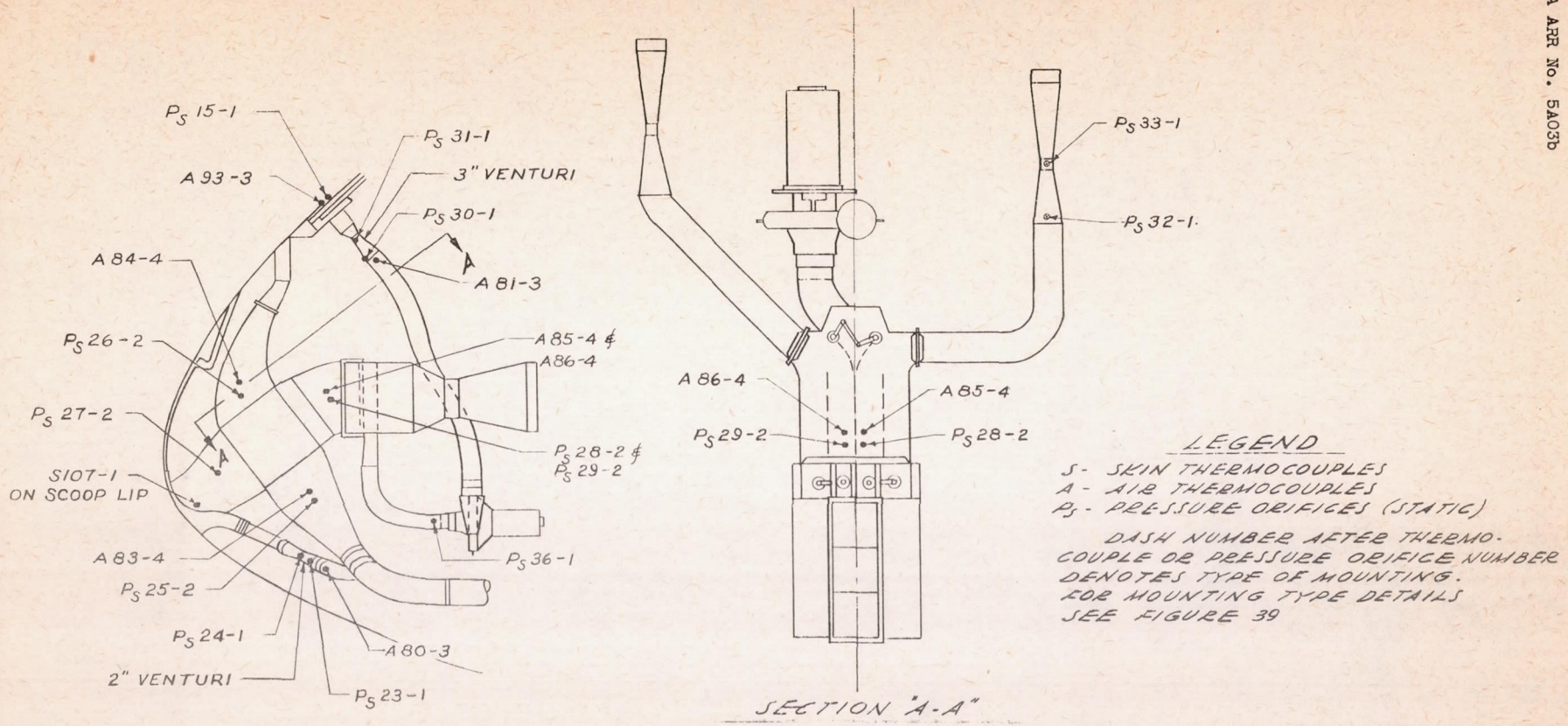
LEGEND

A- AIR THERMOCOUPLES
S- SKIN THERMOCOUPLES

DASH NUMBER AFTER THERMOCOUPLE NUMBER DENOTES TYPE OF MOUNTING

FOR MOUNTING TYPE DETAILS SEE FIGURE 39

FIGURE 37 - THERMOCOUPLE LOCATIONS FOR FIN - C-46 AIRPLANE



LEGEND

S - SKIN THERMOCOUPLES
A - AIR THERMOCOUPLES
P_S - PRESSURE ORIFICES (STATIC)

DASH NUMBER AFTER THERMOCOUPLE OR PRESSURE ORIFICE NUMBER DENOTES TYPE OF MOUNTING. FOR MOUNTING TYPE DETAILS SEE FIGURE 39

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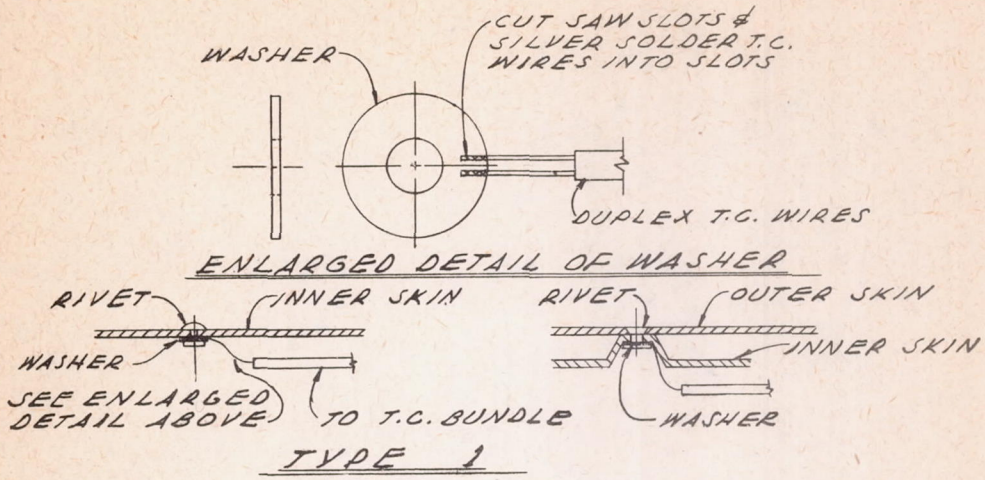
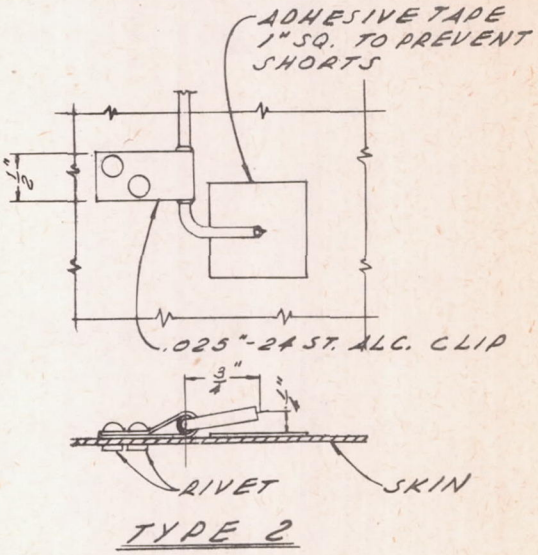
FIGURE 38 - THERMOCOUPLE AND PRESSURE-ORIFICE LOCATIONS FOR SECONDARY HEAT-EXCHANGER INSTALLATION IN NOSE OF C-46 AIRPLANE

A-16



THERMOCOUPLES



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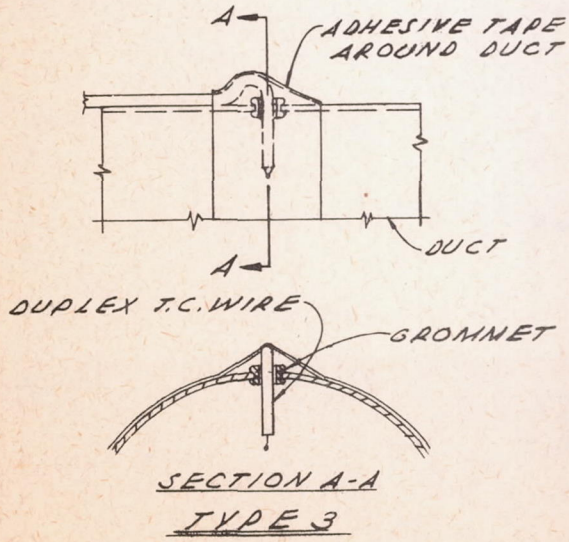

Figure 39(a to c).- Types of thermocouple and pressure orifice installations used to determine performance of ice-prevention equipment of the C-46 airplane.

A-16

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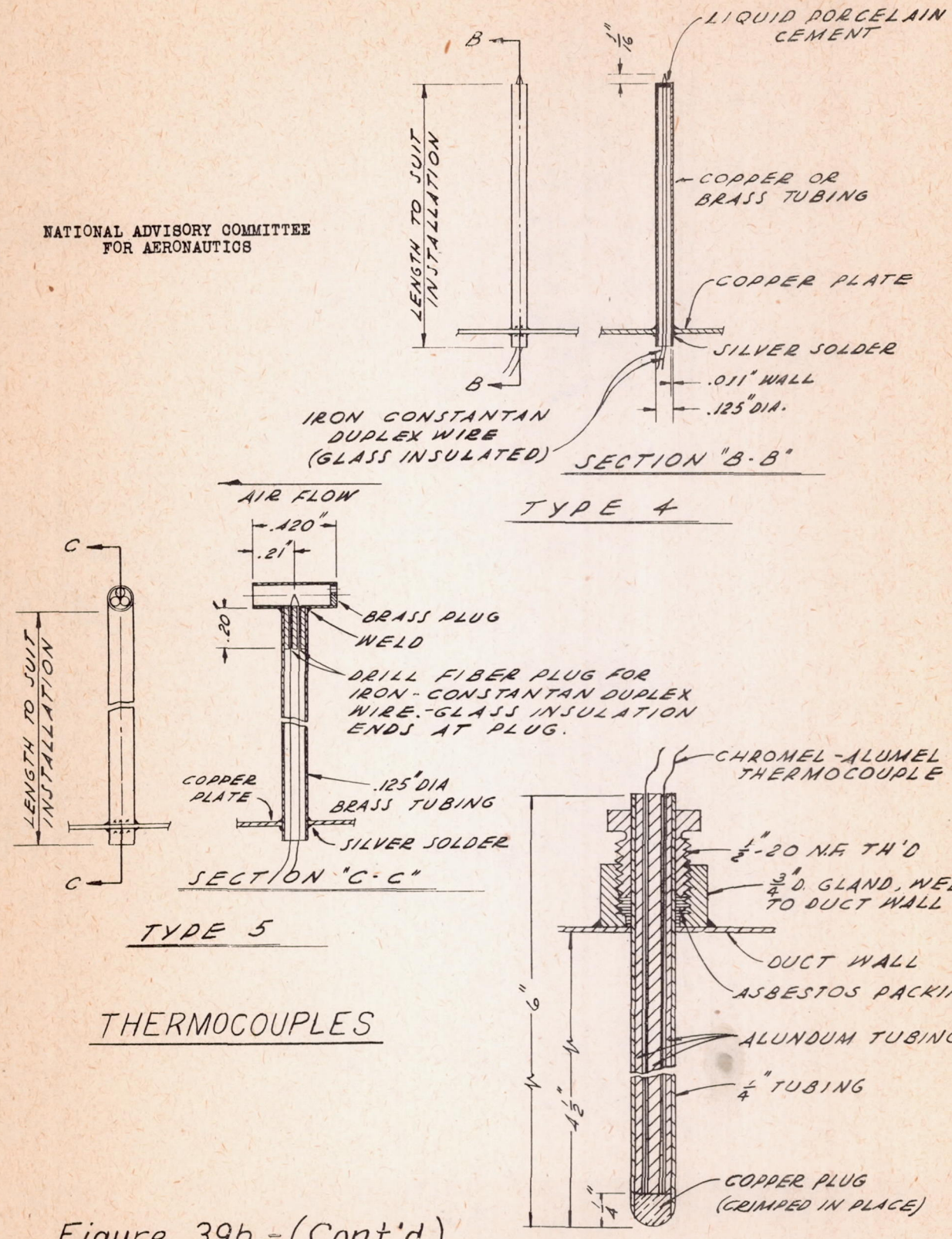
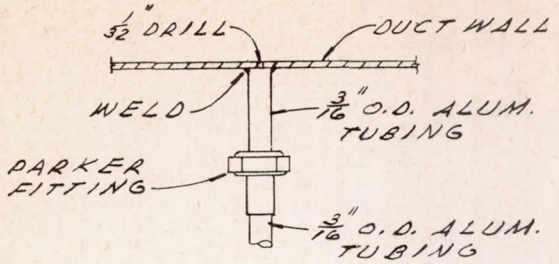


THERMOCOUPLES

Figure 39b.-(Cont'd.)

A-16



TYPE 1

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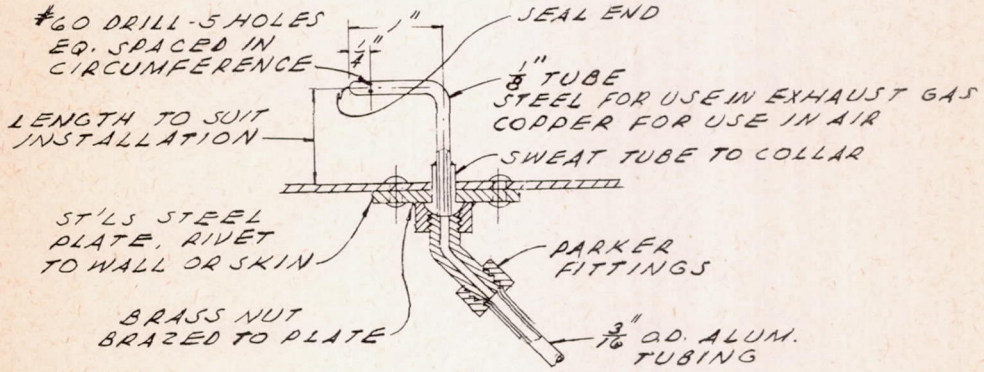
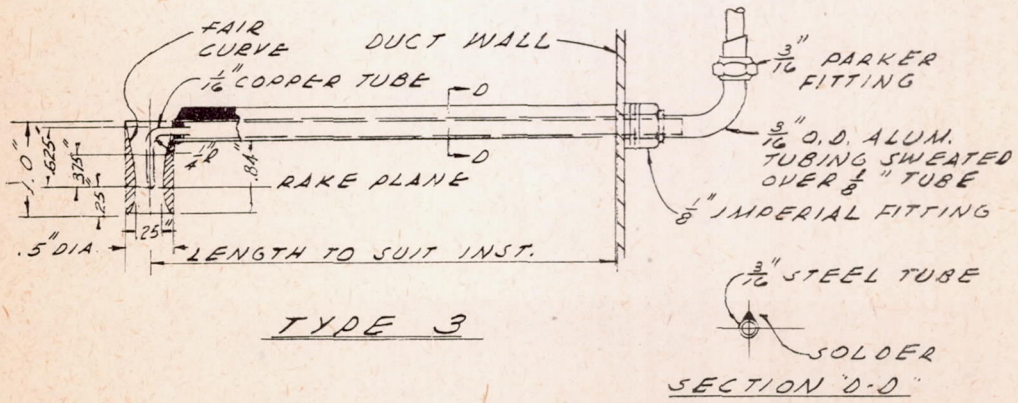


TYPE 2

PRESSURE ORIFICES



TYPE 3

Figure 39c.-(Concl'd.)

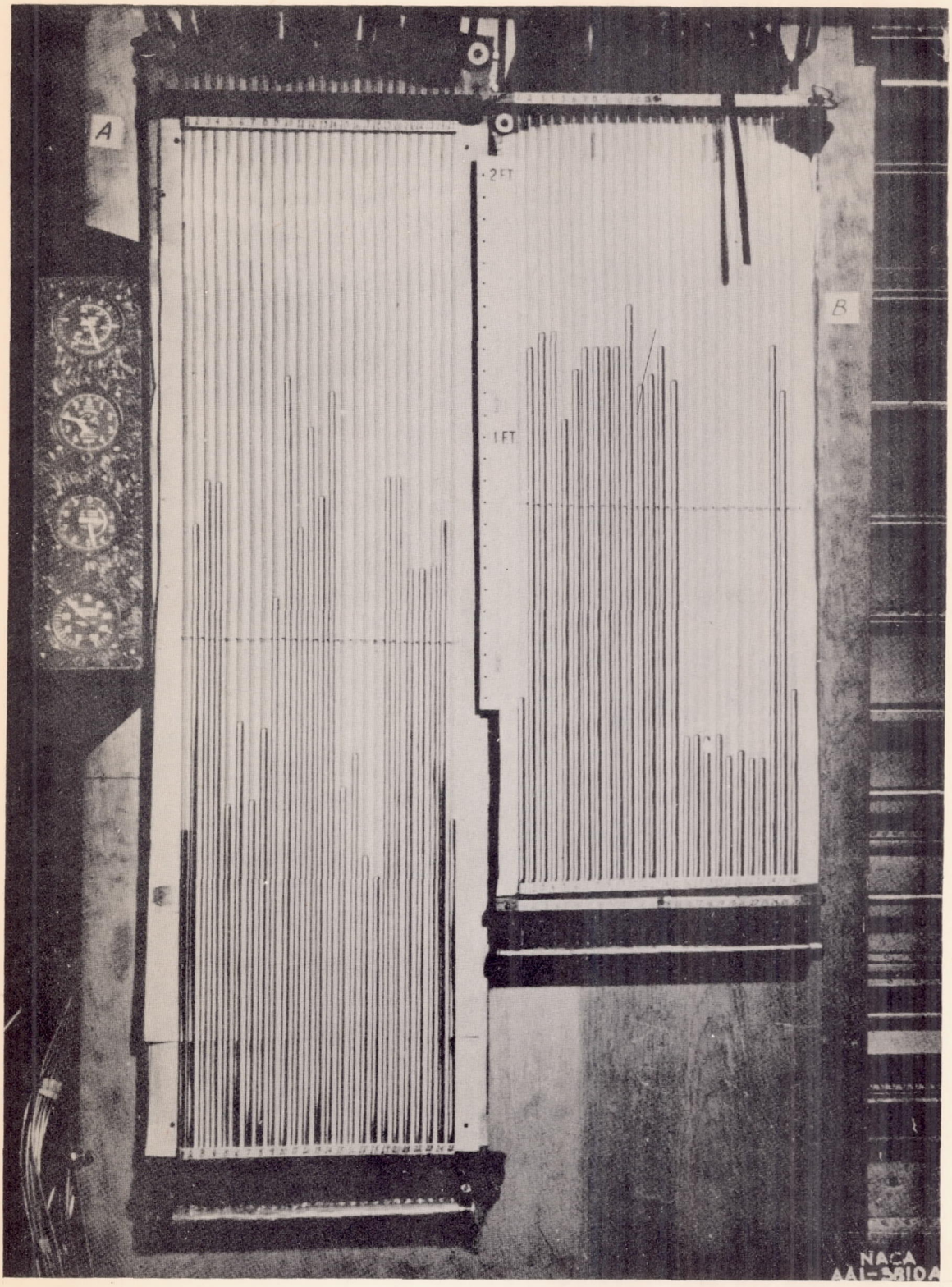


Figure 40.- Alcohol manometer installed in cabin of C-46 airplane for indicating pressure differentials in performance tests of thermal ice-prevention equipment.

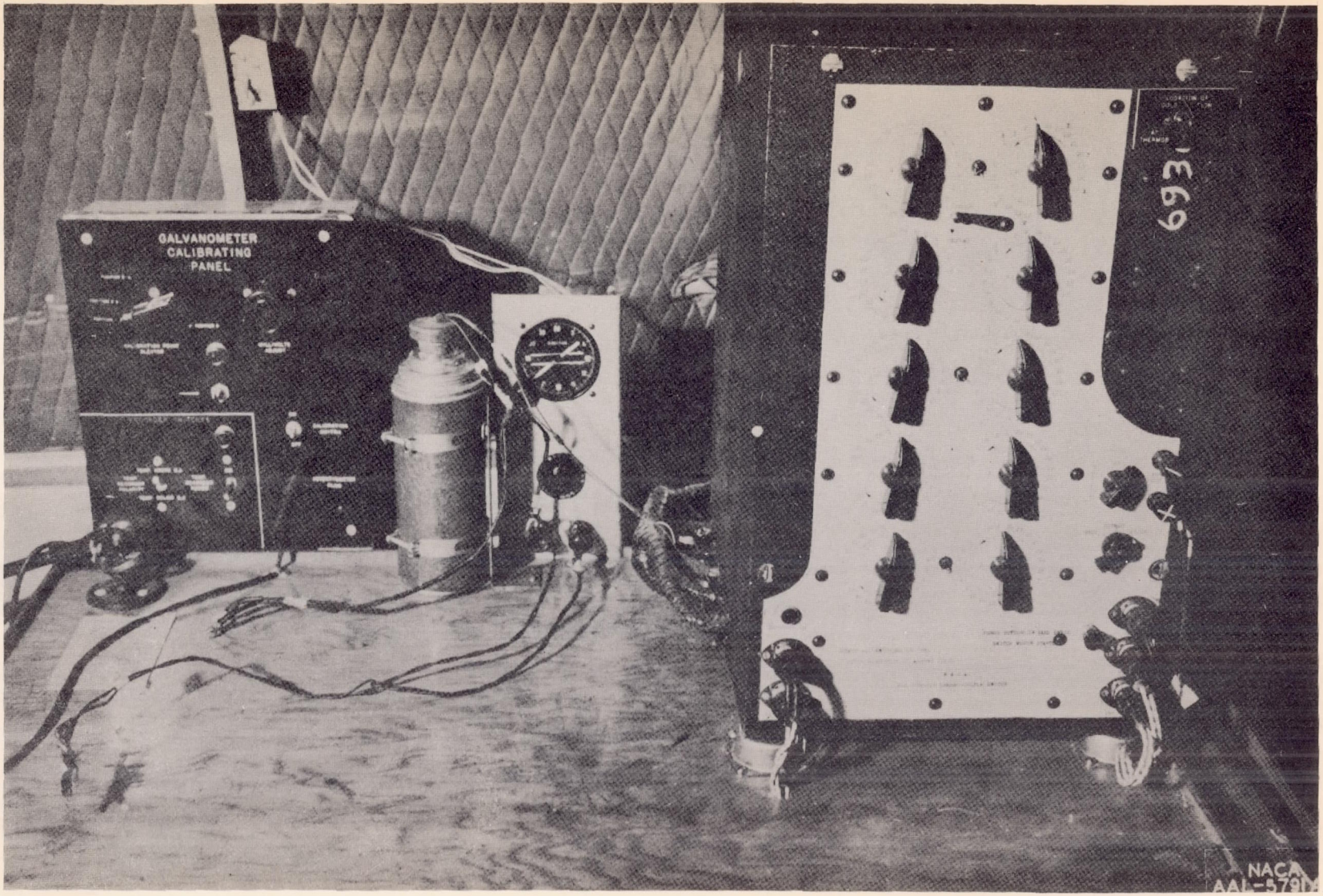
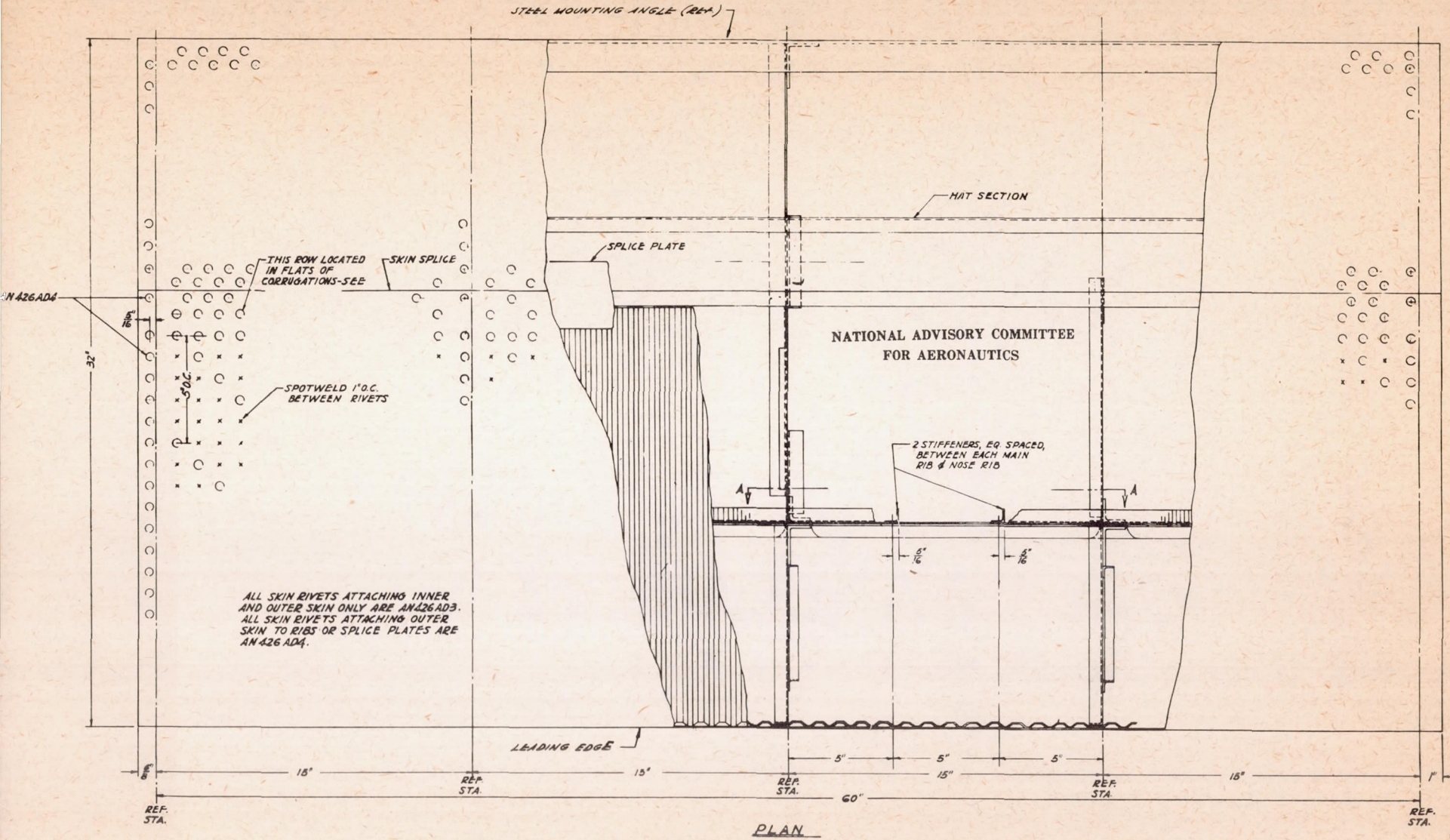


Figure 41.- Thermocouple selecting unit and recorder installed in C-46 airplane for performance tests of thermal ice-prevention equipment.



PLAN

FIGURE 42a,b — ASSEMBLY OF WING OUTER PANEL LEADING EDGE STATIC TEST SPECIMEN, C-46 AIRPLANE

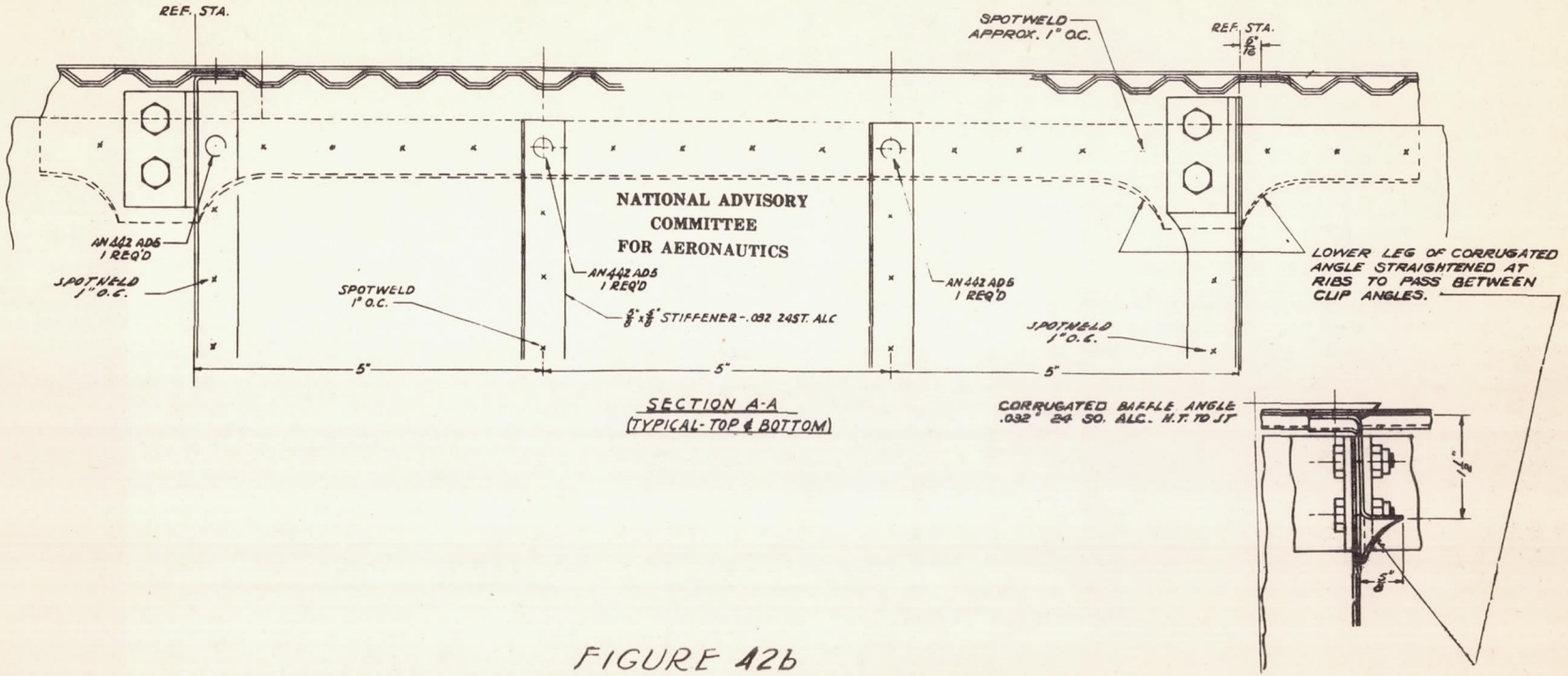

REF. STA.
SPOTWELD APPROX. 1" OC.
REF. STA.
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
AN 442 AD5 1 REQ'D
SPOTWELD 1" O.C.
SPOTWELD 1" O.C.
AN 442 AD5 1 REQ'D
5/8" x 5/8" STIFFENER - .032 24ST. ALC
AN 442 AD5 1 REQ'D
SPOTWELD 1" O.C.
LOWER LEG OF CORRUGATED ANGLE STRAIGHTENED AT RIBS TO PASS BETWEEN CLIP ANGLES.
5"
5"
5"
SECTION A-A
(TYPICAL-TOP & BOTTOM)
CORRUGATED BAFFLE ANGLE .032" 24 SO. ALC. N.T. TO ST

FIGURE 42b

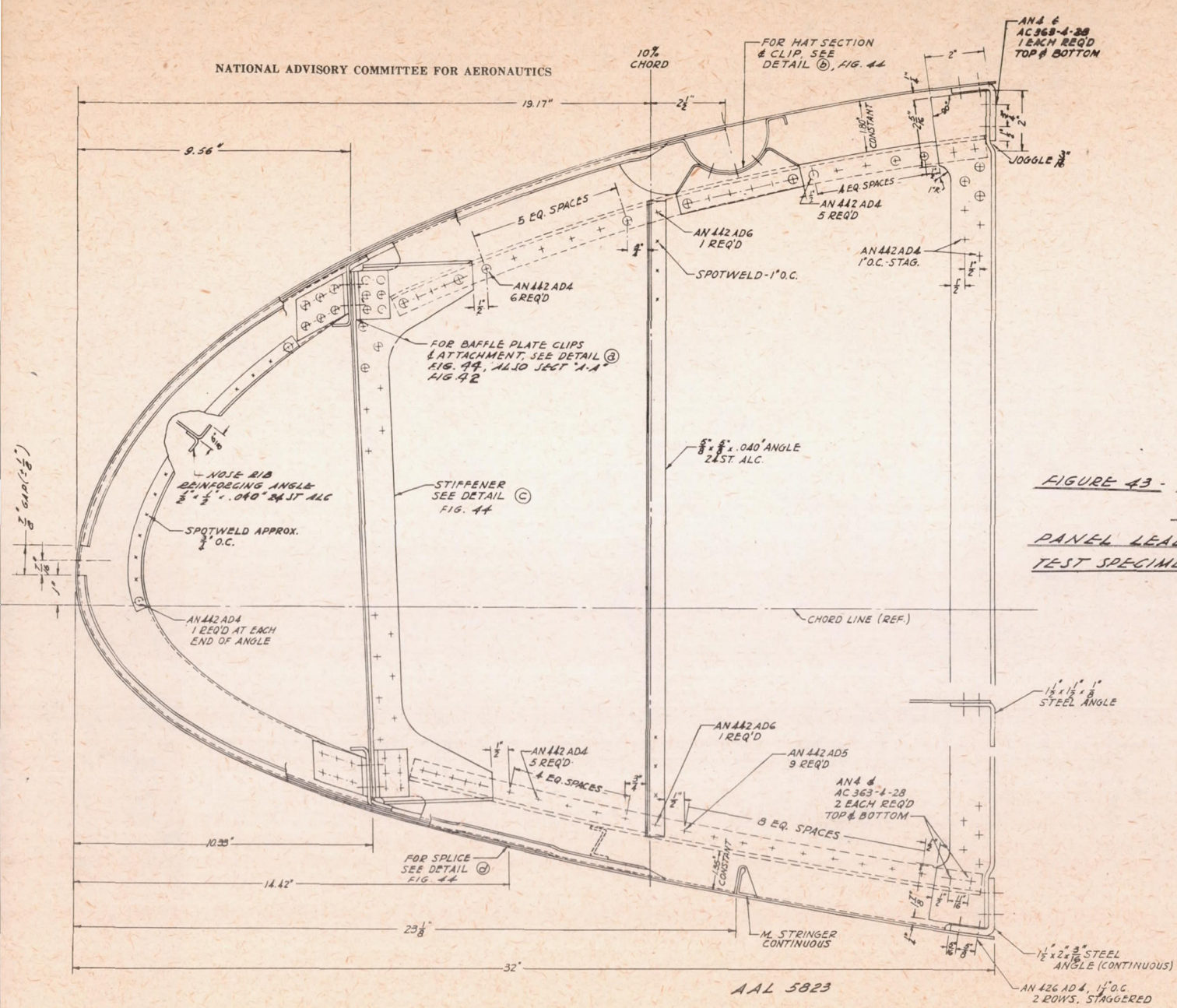
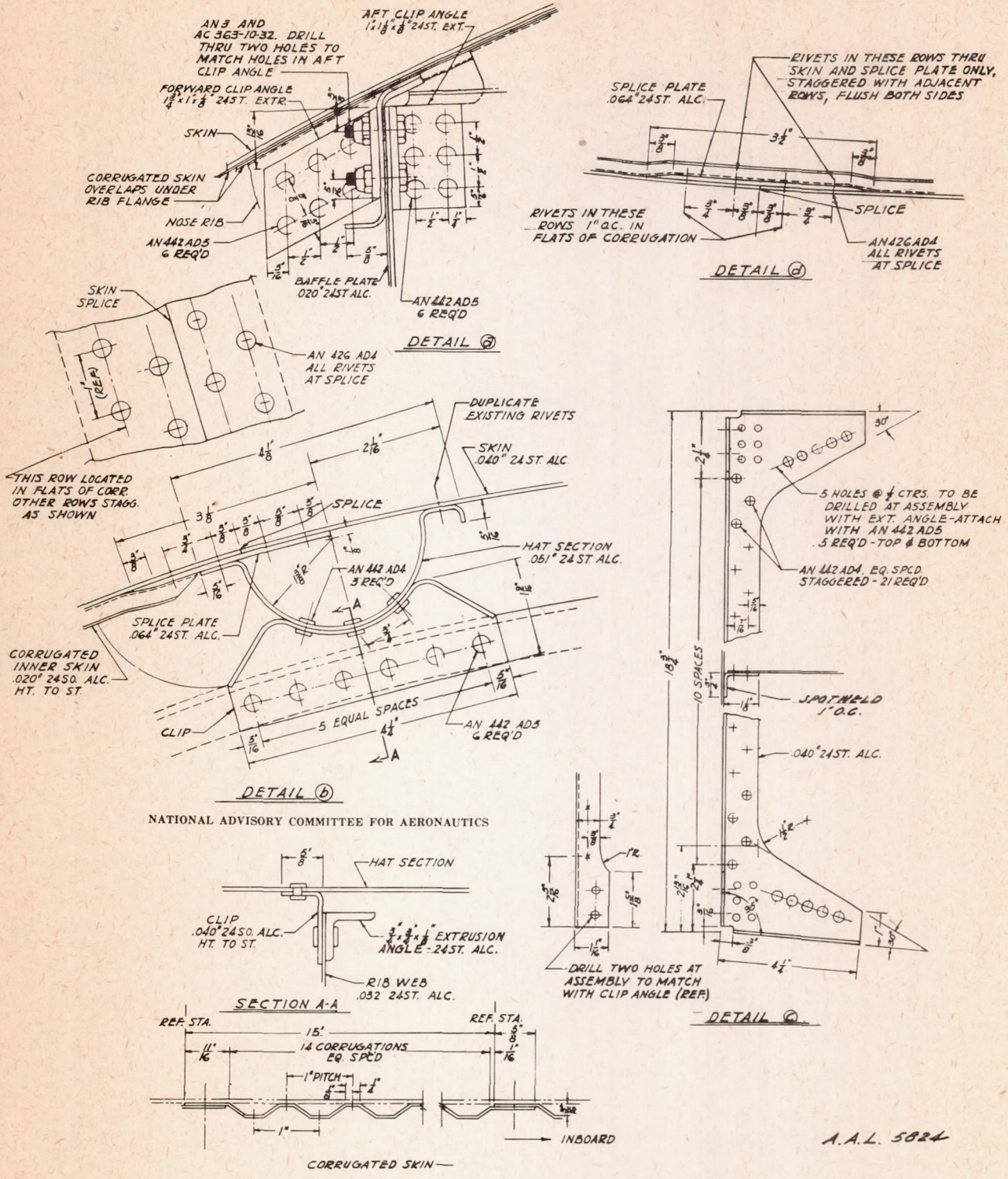


FIGURE 43 - MAIN RIB ASSEMBLY FOR WING OUTER PANEL LEADING EDGE, STATIC TEST SPECIMEN, C-46 AIRPLANE

A-16



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INNER & OUTER SKIN DETAILS

NOTE: ALL DETAILS SHOWN ARE INDICATED IN FIGURE 43

FIGURE 44. MAIN RIB AND SPLICE DETAILS FOR WING OUTER PANEL LEADING EDGE STATIC TEST SPECIMEN C-46 AIRPLANE

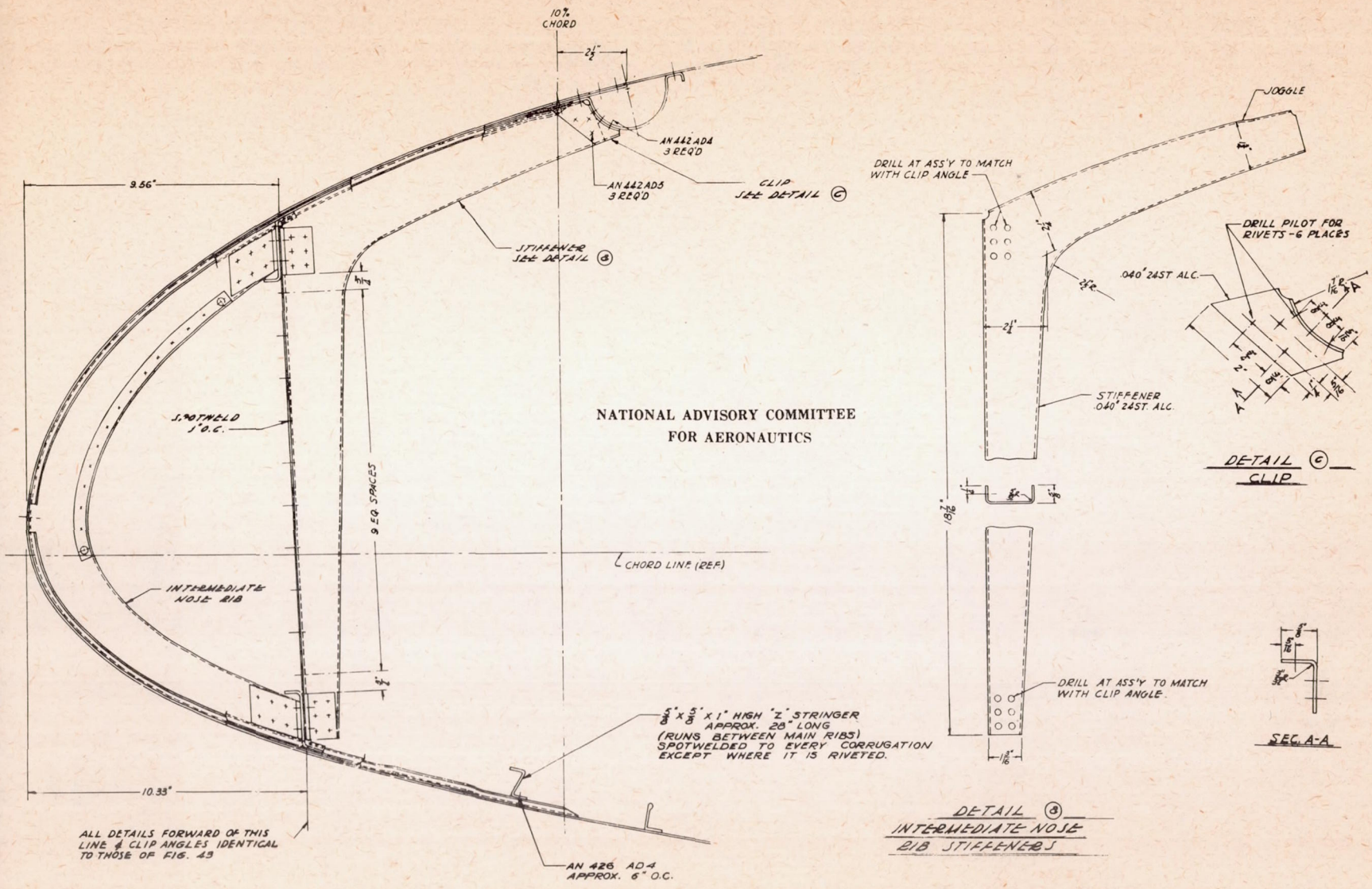

FIGURE 45 - INTERMEDIATE RIB DETAILS FOR WING OUTER PANEL LEADING EDGE STATIC TEST SPECIMEN C-46 AIRPLANE

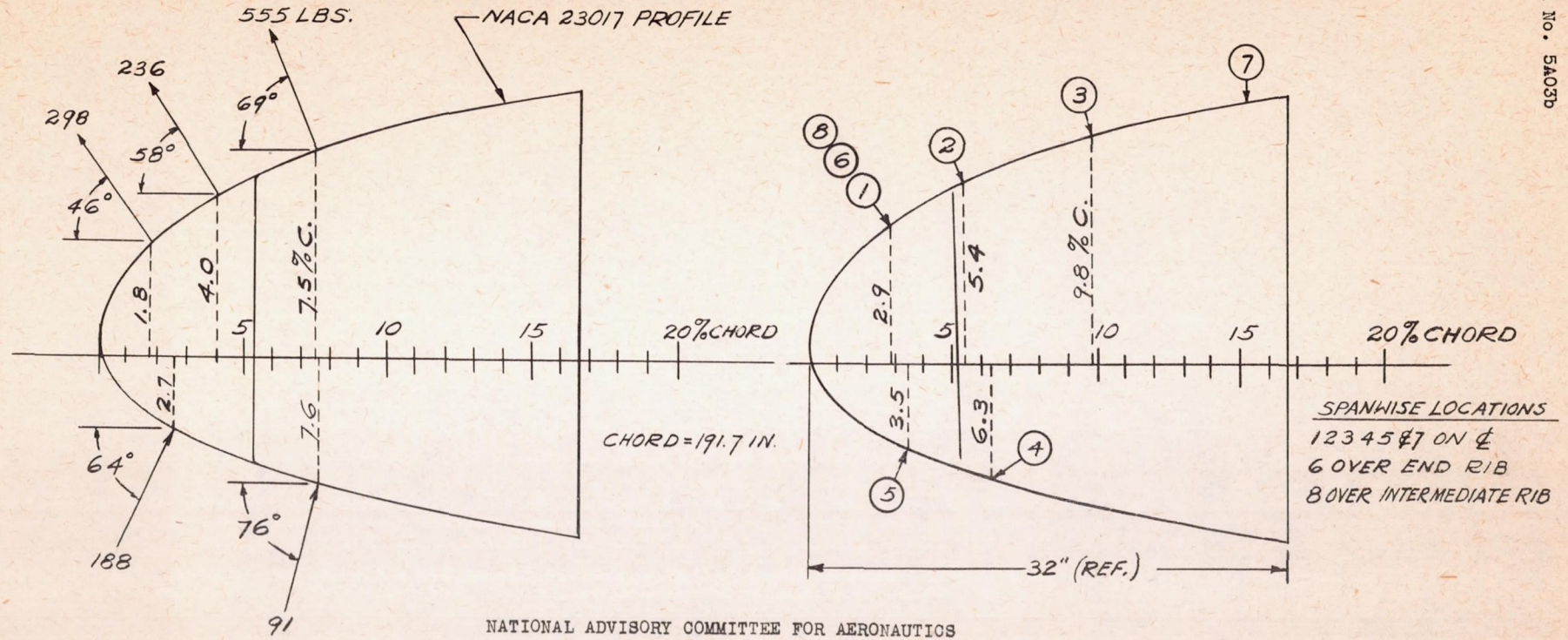

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FIGURE 46.- MAGNITUDE, DIRECTION, AND LOCATION OF ULTIMATE LOADS FOR 15-INCH SECTION OF STATIC TEST SPECIMEN OF C-46 WING OUTER-PANEL LEADING EDGE.

FIGURE 50.- LOCATIONS OF AMES DEFLECTION DIALS FOR STATIC TESTS OF C-46 WING LEADING-EDGE SPECIMEN.

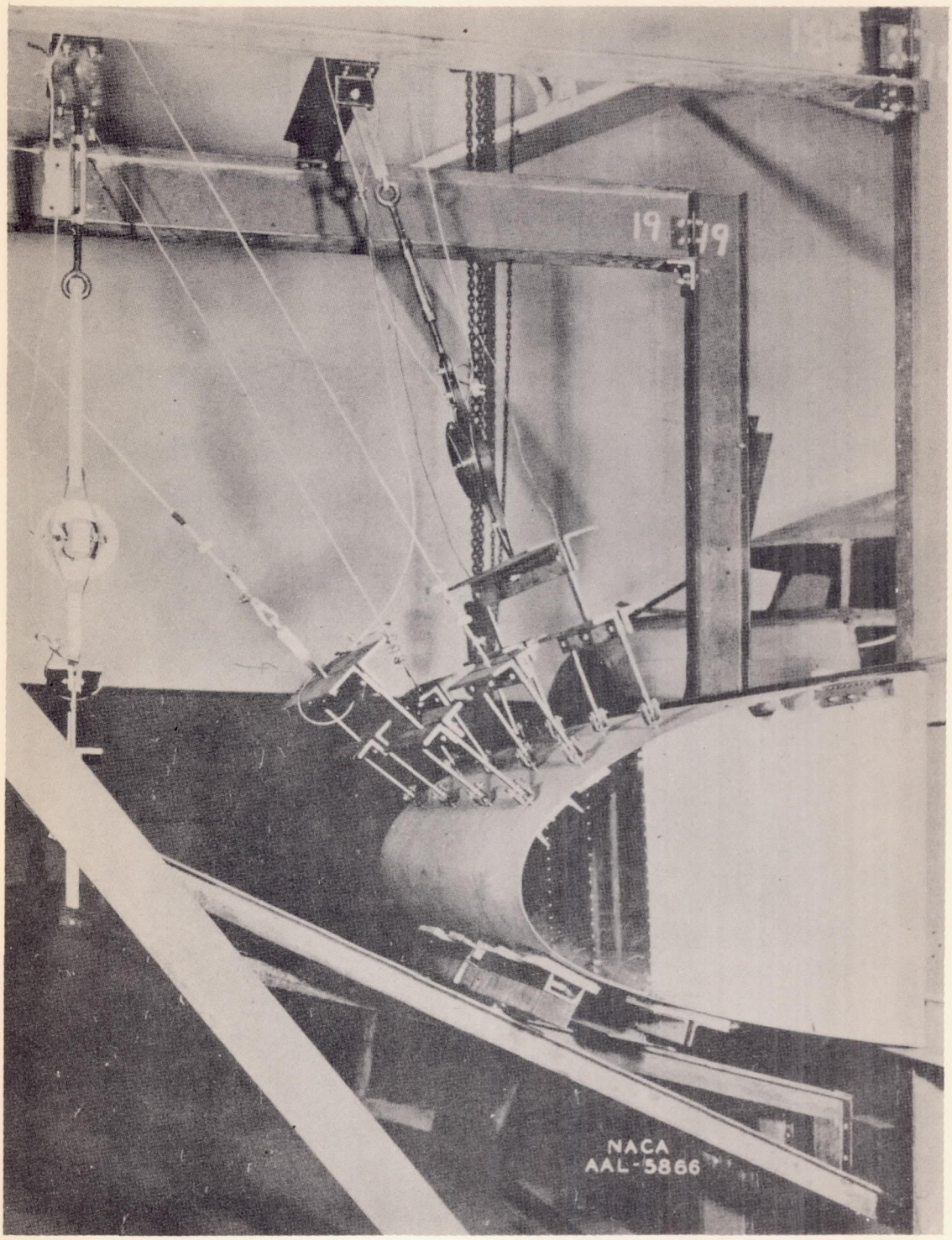

Figure 47.- Side view of system for applying static loads to structural test specimen of C-46 wing outer-panel leading edge.

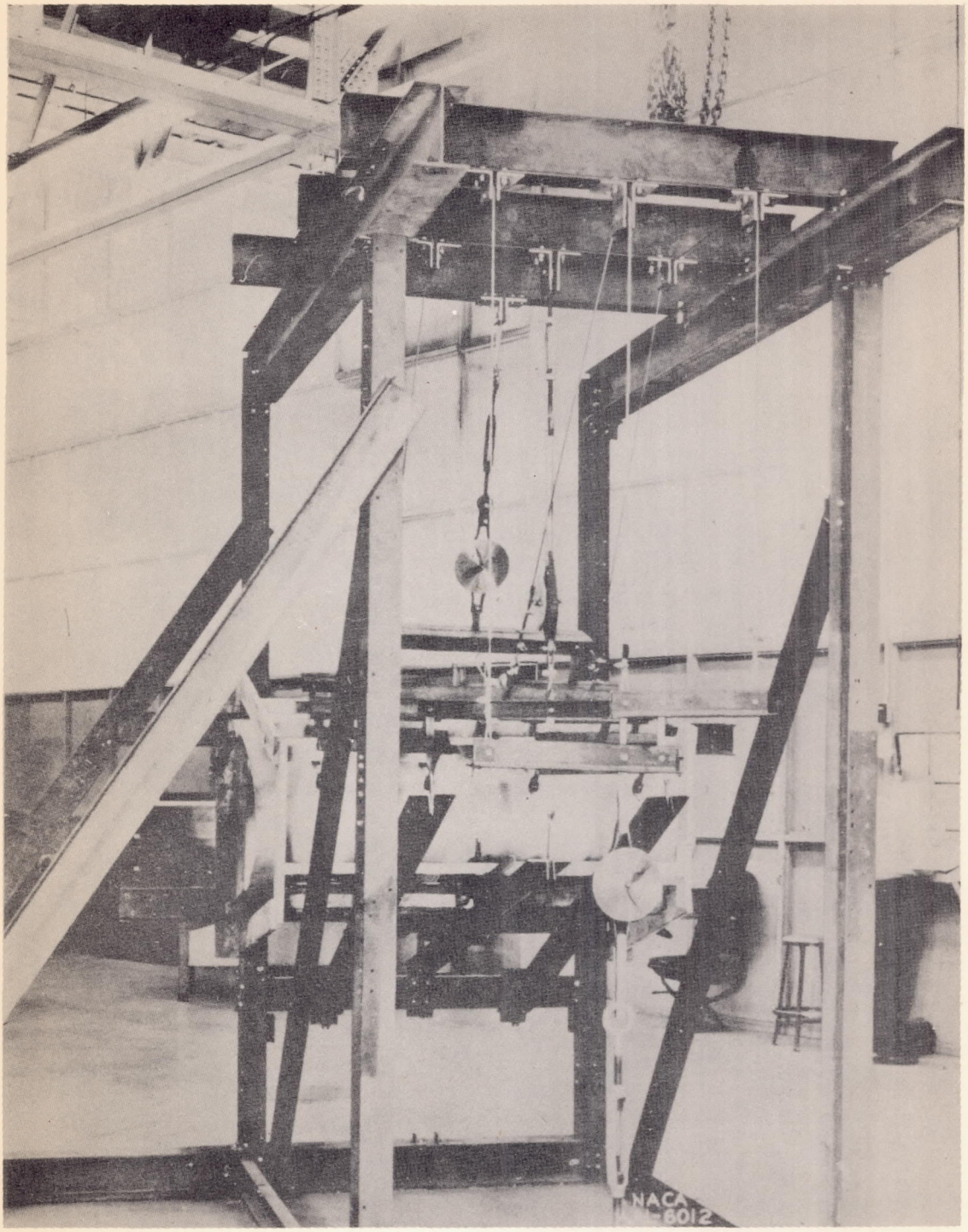

Figure 48.- Front view of system for applying static loads to structural test specimen of C-46 wing outer-panel leading edge.

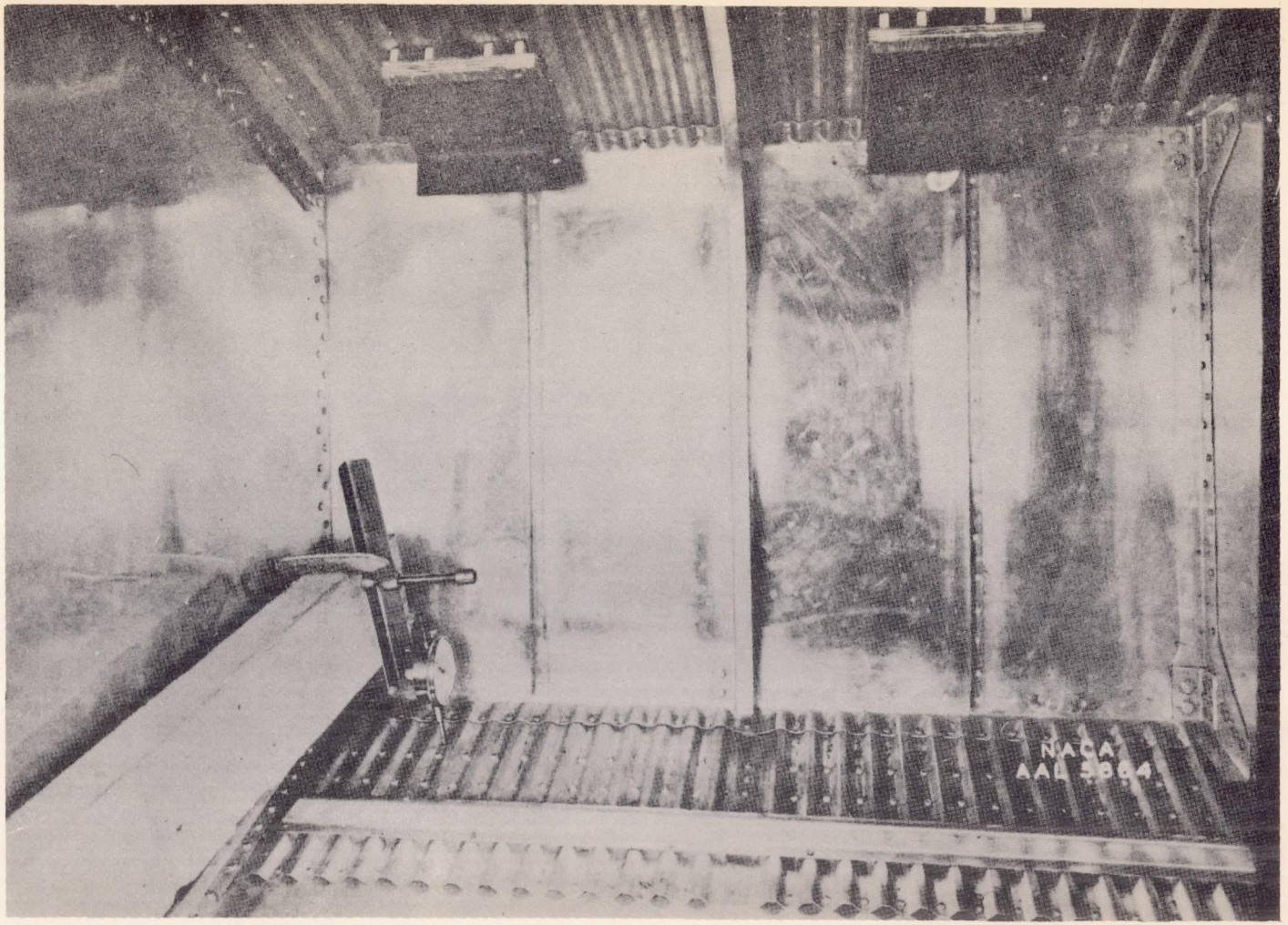

Figure 49.- Rear view of C-46 wing leading-edge static test specimen showing channel and wood-block arrangement for distributing tension load to corrugations.

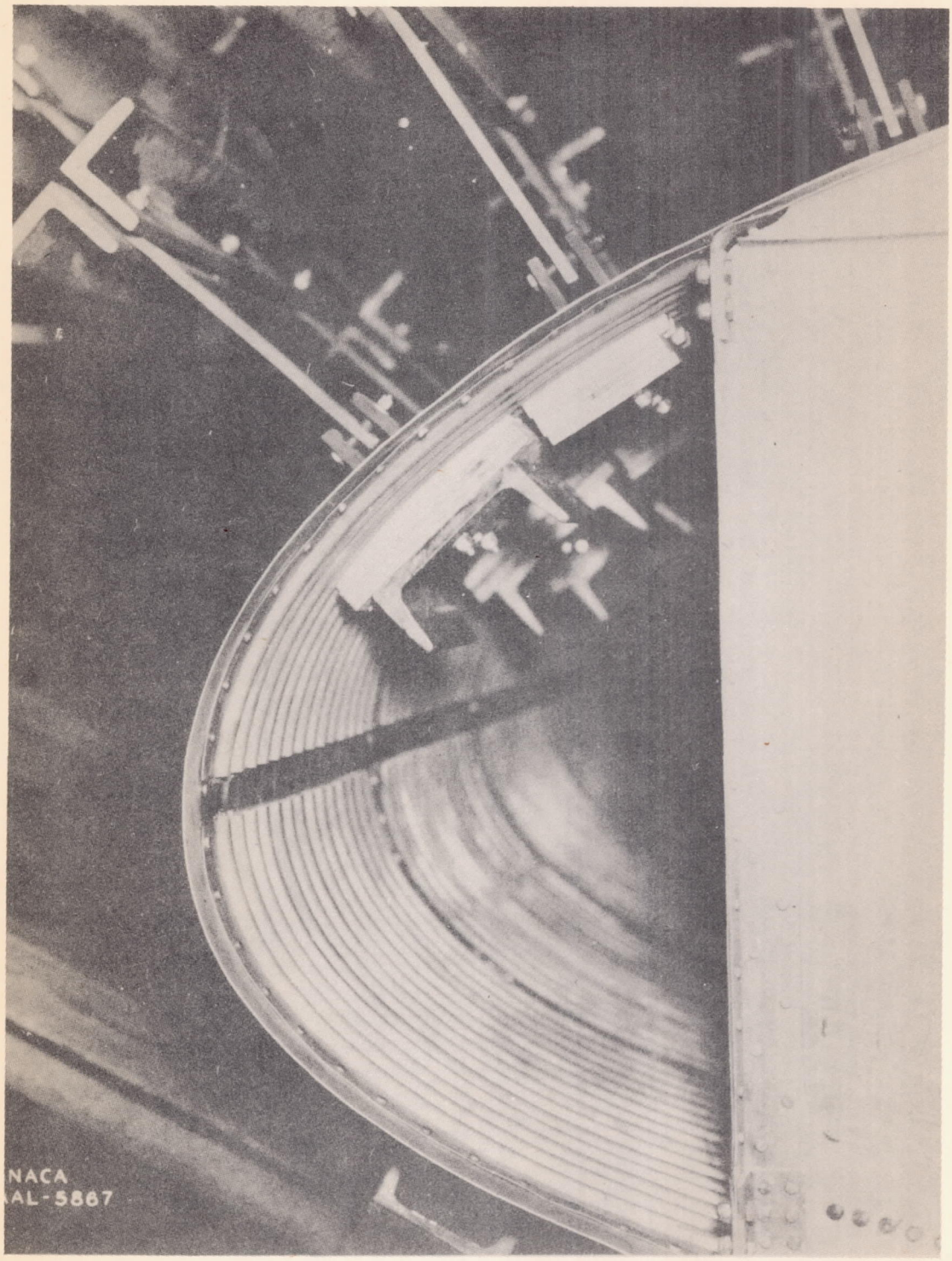

Figure 51.- Static test specimen of C-46 wing outer-panel leading edge as altered for test number 5.

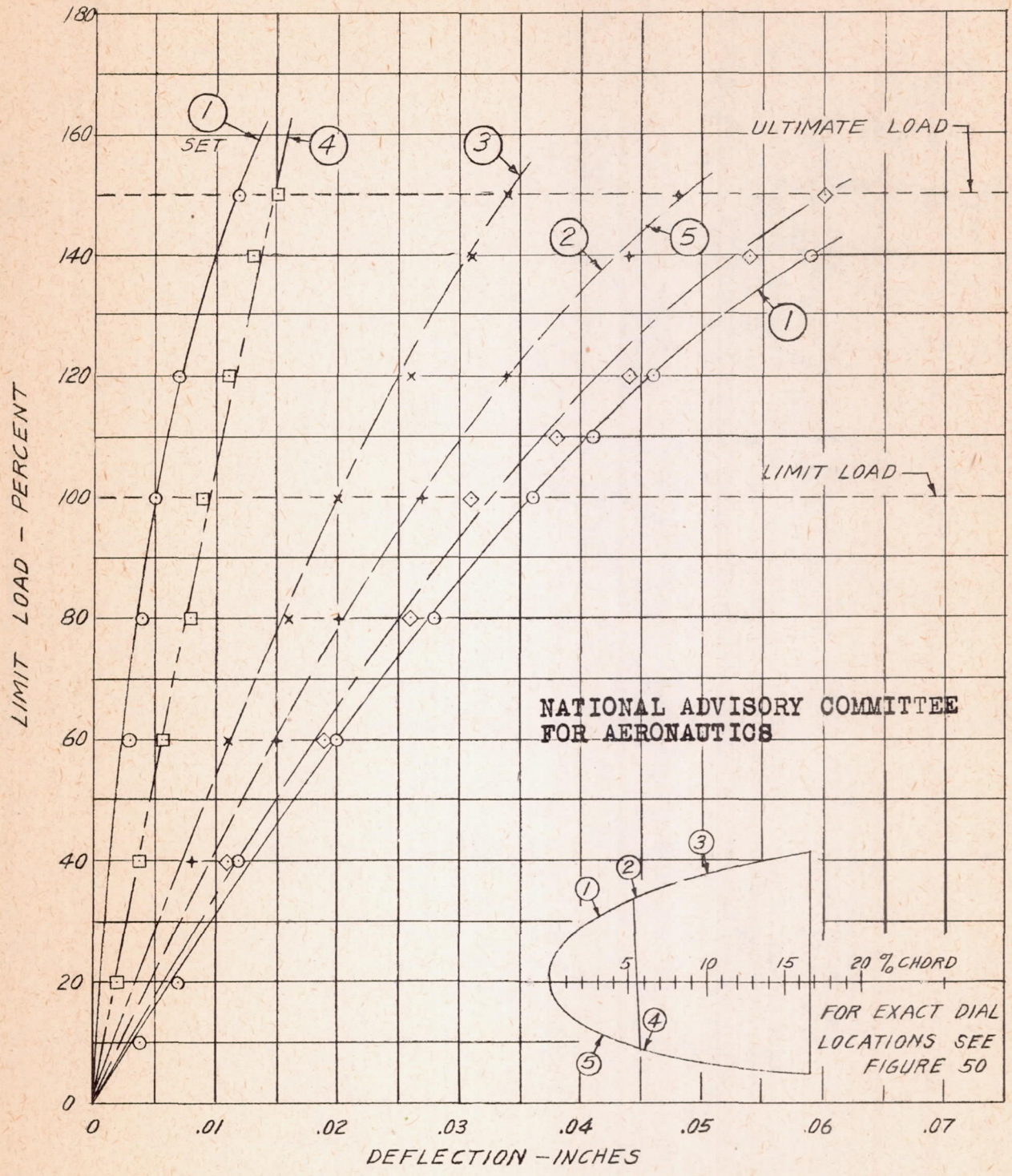

FIGURE 52.- DEFLECTIONS INDICATED BY AMES DIALS DURING STATIC TEST NUMBER ONE, C-46 WING OUTER-PANEL LEADING-EDGE SPECIMEN.

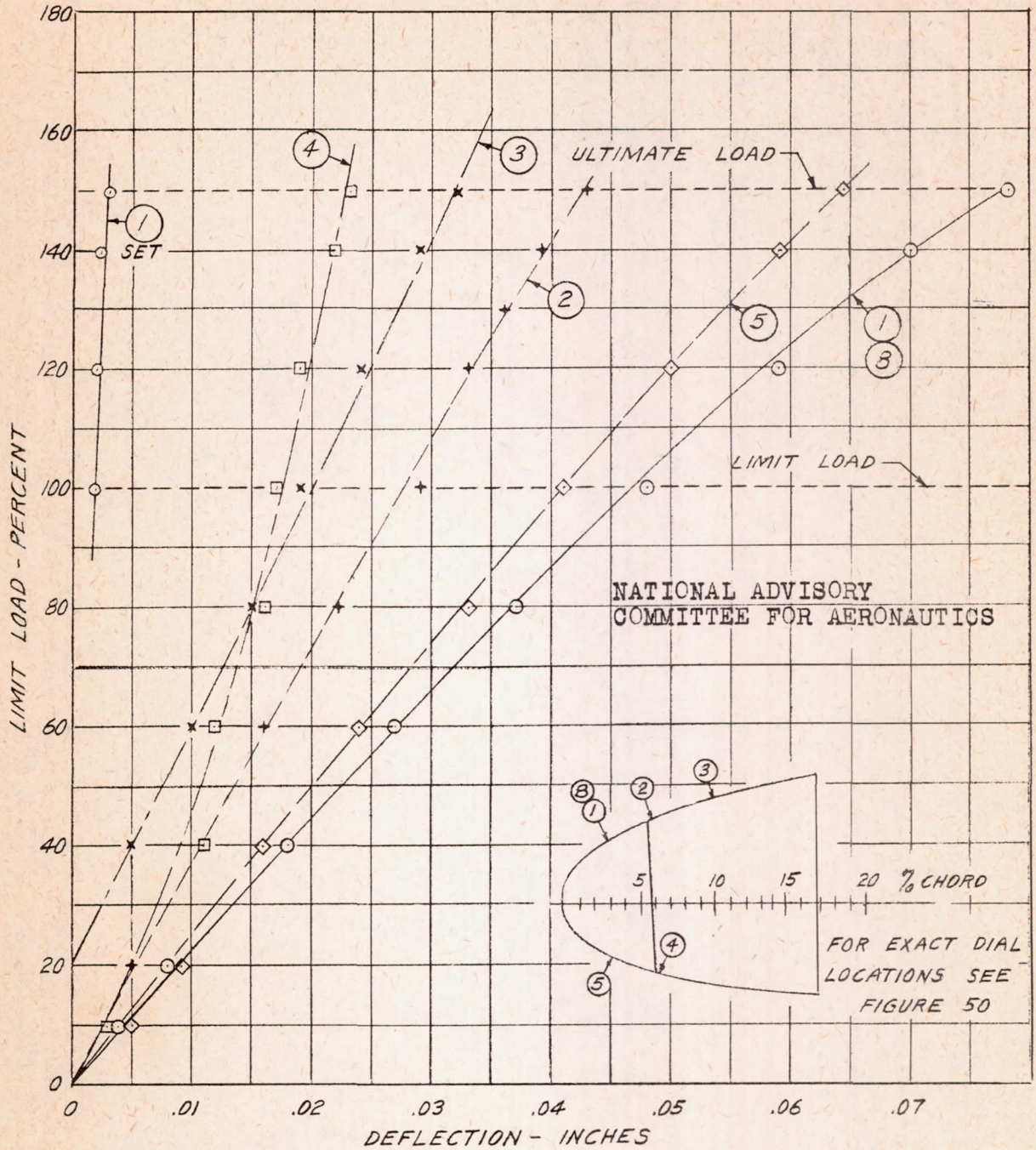

FIGURE 53.- DEFLECTIONS INDICATED BY AMES DIALS DURING STATIC TEST NUMBER FOUR, C-46 WING OUTER-PANEL LEADING-EDGE SPECIMEN.

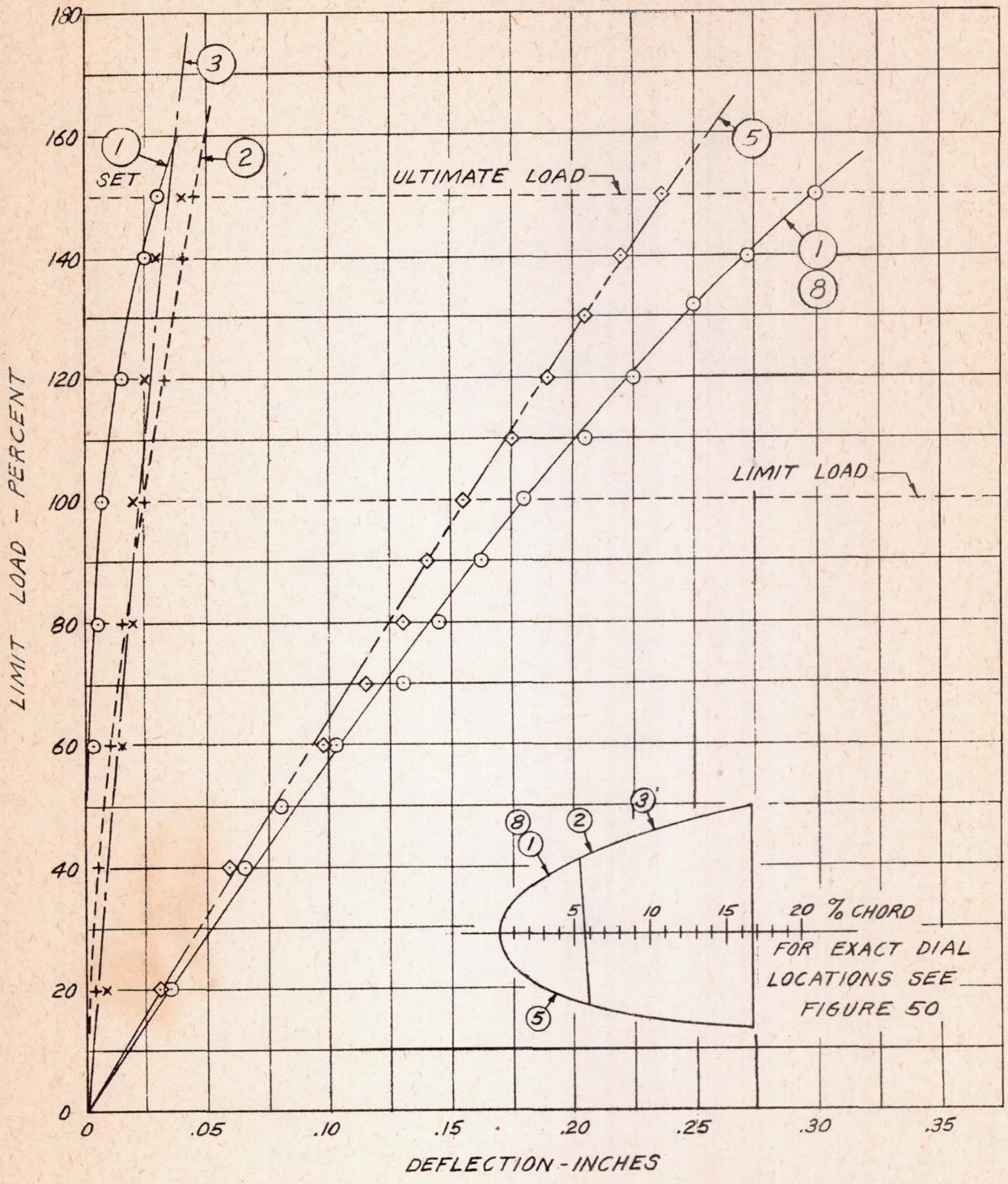

FIGURE 54.- DEFLECTIONS INDICATED BY AMES DIALS DURING STATIC TEST NUMBER FIVE, C-46 WING OUTER-PANEL LEADING-EDGE SPECIMEN.